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TO: Chief, Nuclear Power Division
Office of the Chief of Engineers
Washington 25, D.C.

MONTHLY REPORT.

ENGINEERING STUDY
OF MULTIPURPOSE ENGINE
AND FUEL SYSTEM
FOR MANNED LUNAR BASES,

Contract DA-49-129-ENG (NASA)-1

Westinghouse Electric Corporation
Astronuclear Laboratory

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LUNAR BASE
ENERGY SYSTEMS
PROJECT

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I. SUMMARY

A. GENERAL

Major effort during the month has been expended in conducting state-of-the-art review on the potential fuels and regeneration processes and in obtaining current performance information on energy conversion devices. Several systems have been designed for the shelter power system as a method of establishing performance and system parameters that may be utilized in the future selection of a preferred concept.

The total program schedule is outlined. With few exceptions all areas are proceeding as planned, and no major roadblocks have been uncovered which would materially alter the phase of action.

One critical area requires the establishment of a solid guideline which could not be reasonably derived by Westinghouse and was not adequately covered during the NASA-OCE-Westinghouse visit to the Boeing Company. In the Boeing concept of a lunar base, very limited use is made of lunar roving vehicles. Many other investigators have proposed substantially greater mobility requirements. It is therefore necessary that NASA-OCE and Westinghouse derive a "mission profile" is required for the various lunar roving vehicles which can be utilized to evaluate the vehicle power system concepts and their effect on fuel logistic systems.
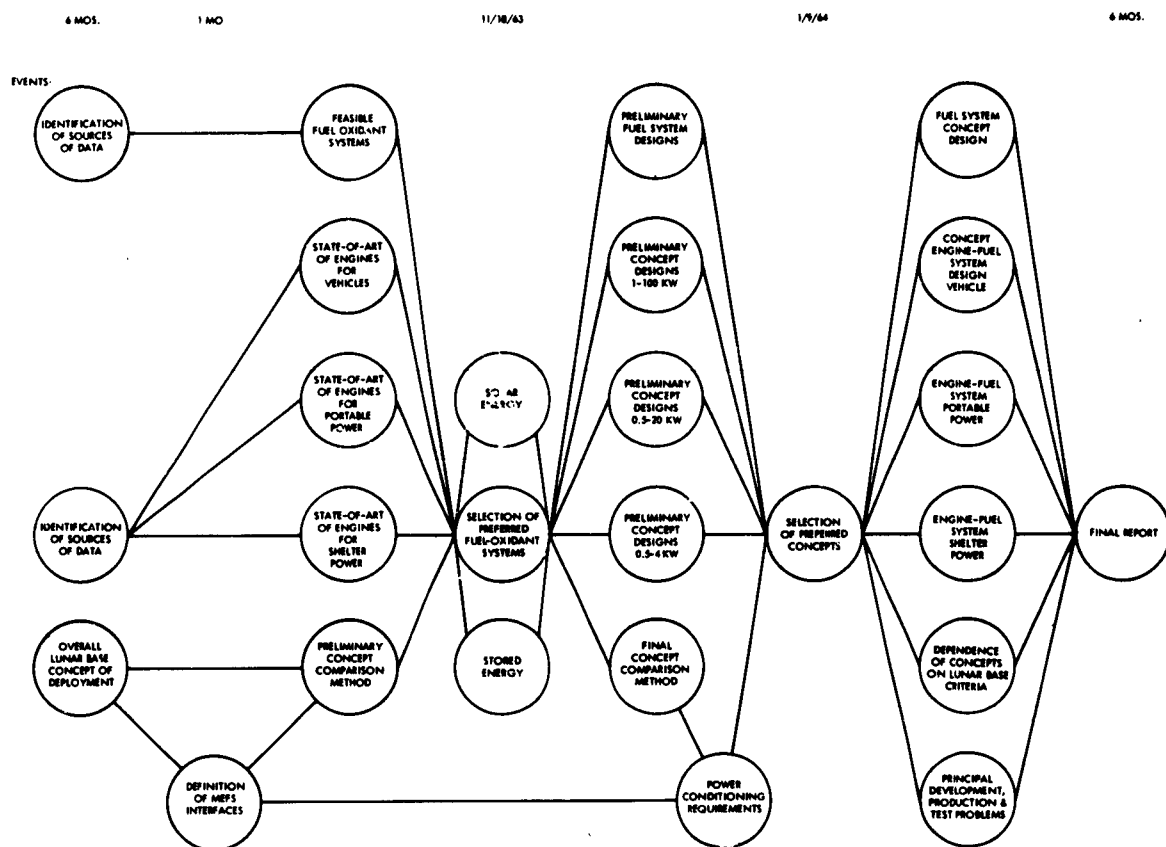
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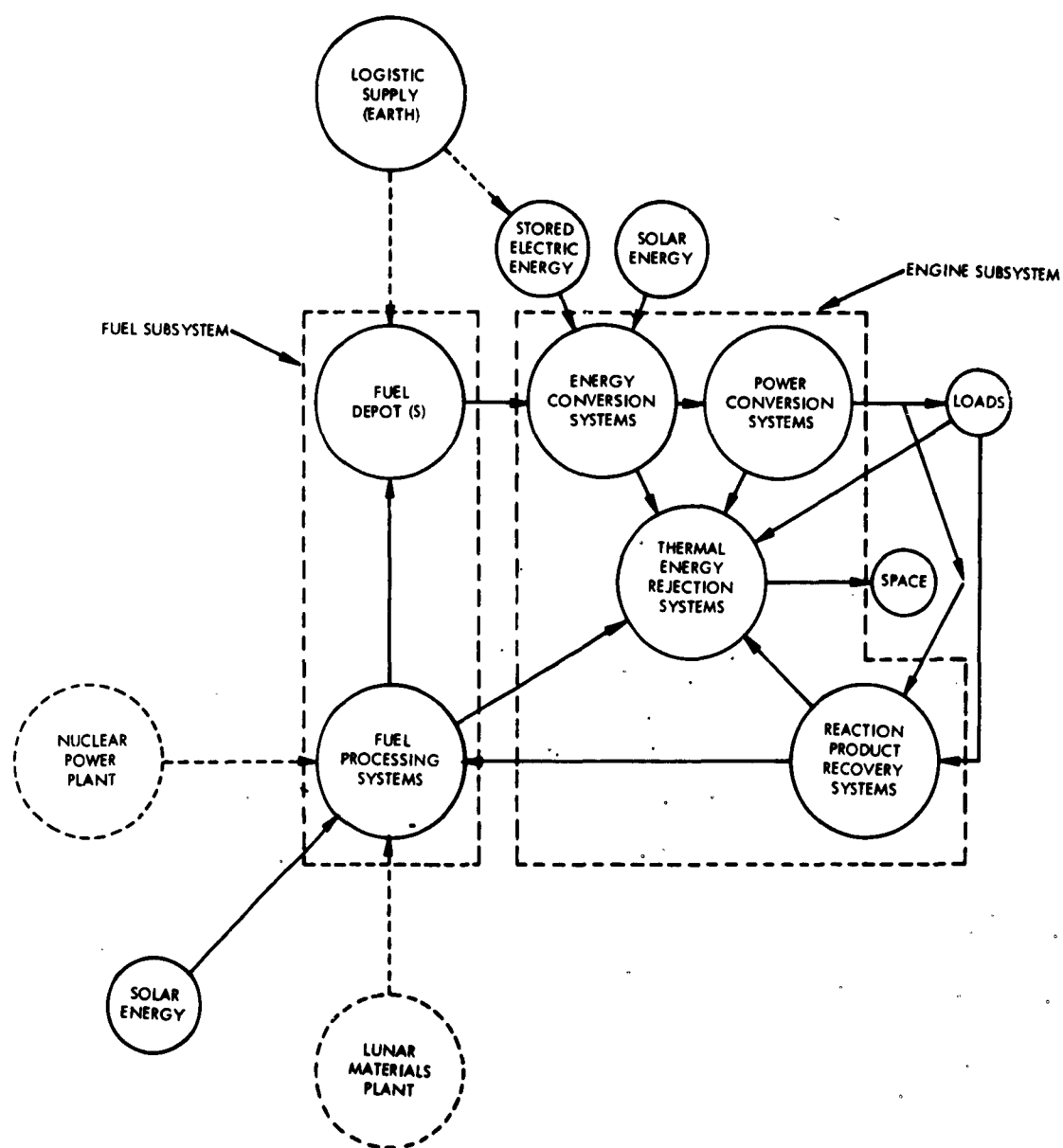
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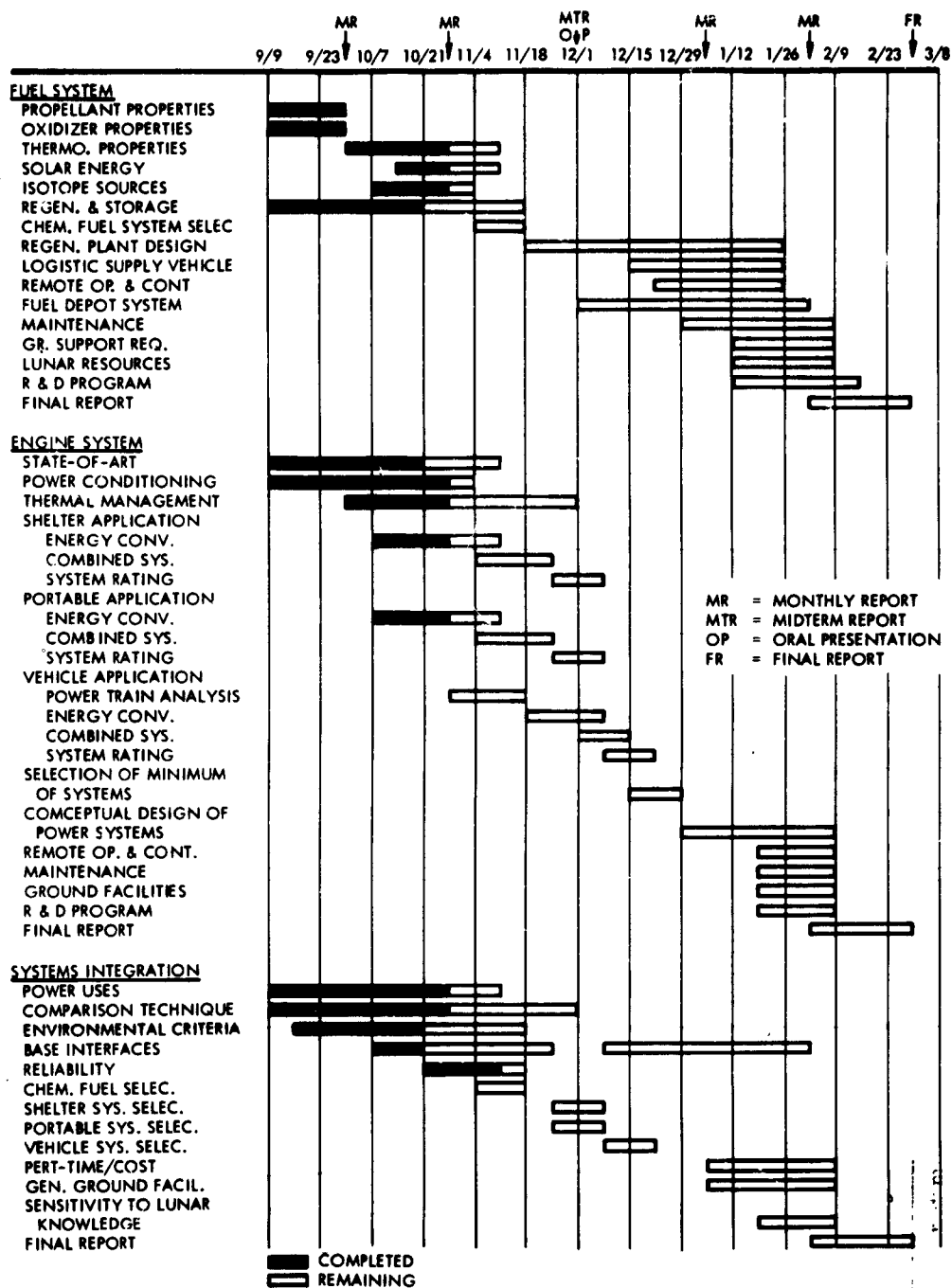
Figure 1-3. Contract DA 49-129-ENG-(NASA)-1(B) Multi-Purpose Engine and Fuel System Study Schedule

B. SYSTEMS INTEGRATION

Effort during October was concentrated in the following areas:

Power Uses

Environmental Criteria

Interface Considerations

Reliability

Comparison Technique

The power use effort in September specified a certain load profile against which systems could be compared. This profile included yearly totals of engine installed capacity and power usage. A visit was made to the Boeing Company, to obtain their full load profile information from the lunar base concept study prior to its publication. Although Boeing's concepts of installed capacity are in fair agreement with our September report, the difference in assumed base activities results in the majority of power being supplied by central station power under the Boeing concept. It is concluded that the model presented by Westinghouse will provide a more realistic base for evaluating multipurpose engine and fuel concepts.

The environmental requirements as stated in the letter request were reviewed, and some engineering criteria were evolved. Criteria in the meteoroid area are presented. In November, all criteria required as a result of environmental guidelines will be presented.

The area of launch and landing consideration (interface) was also examined. It appears that there are no critical (affects concept comparison) criteria involved. This effort will be summarized and reported next month.

Effort in the area of reliability has resulted in a statement of criteria against which engine and regeneration systems will be compared. Component mean time between failure has been used as a primary measurement of reliability.

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A preliminary comparison of several typical systems is presented to demonstrate the technique that will be used in comparing systems. As expected, those systems using a solar energy source appear extremely attractive. Subsequent reiteration of this type of analysis will be employed as more definitive information is made available on specific engine performance, reliability, development costs, etc.

A typical "PERT type" system development chart is shown in Figure 1-4. This chart will serve as a guide for estimating development status and cost of the compared systems. It should be noted that devices for which materials feasibility has not been demonstrated at this time are not considered to be high candidates to meet the 1968 Flight-Test guideline.

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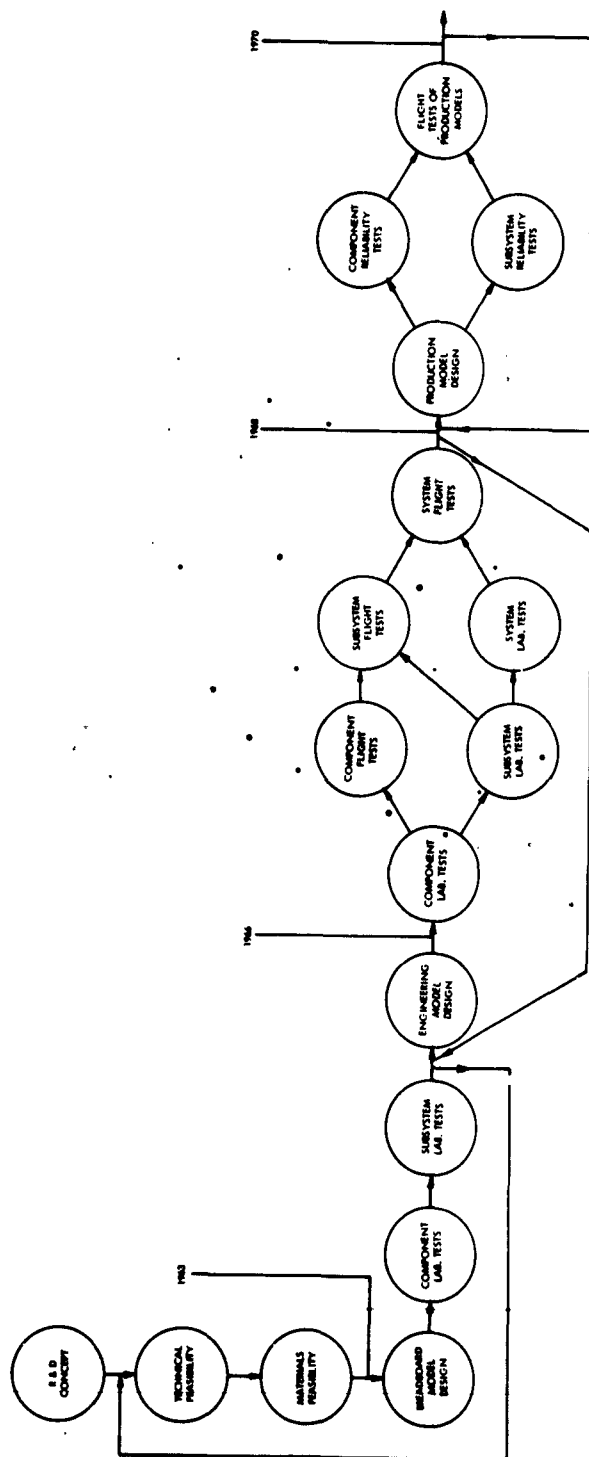
Figure 1-4. System Development Chart

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C. SUPPLIER VISITS

A substantial amount of time was spent this month visiting companies presently developing engines with a potential lunar application. Pertinent information obtained during these visits is presented in this report.

HEAT REJECTION PERFORMANCE CALCULATIONS

Because of the importance of heat rejection radiators on the surface of the moon, a calculation procedure has been developed which will yield weight and area as a function of a number of relevant variables such as input temperatures, configuration geometry, solar absorbtivity, meteoroid protection, etc. Several cases have been calculated for immediate use, but because of the excessive calculation time required, a computer program is being written to provide the broad range of information needed in a short length of time. The calculation procedure along with some preliminary results is given herein.

SHELTER ENGINE PERFORMANCE

Performance calculations have been made for the shelter power supply using a fuel cell/solar dynamic engine combination and stoichiometric hydrogen and oxygen I. C. engine/solar dynamic engine combination. Details of the calculations and the performance curves are given, but no comparative analysis has been made pending results of the calculations currently being made on several other shelter engine subsystems.

SOLAR PHOTOVOLTAIC CELL AND SOLAR T/E CELL PARAMETRIC PERFORMANCE INFORMATION

A substantial amount of material was generated on solar cells and solar T/E cells. This material will be incorporated in the engine selection analysis during November.

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NOVEMBER EFFORT

During the coming month, the performance of the rest of the candidate engine
subsystems for the shelter will be calculated, and their comparative capability
will be analyzed. In addition, potential engine subsystems for the portable
power supply will be formulated, and their performance will be calculated
and rated. The vehicle effort in November will include the detailed analysis
of the vehicle tractive and auxiliary power characteristics, the formulation
of engine and power train subsystems which will provide these characteris-
tics, and the initiation of performance analysis of these subsystems.

D. SUMMARY OF ENERGY SOURCES (FUELS)

Fourteen combinations of fuels and oxidants were selected for consideration as propellants for the lunar engines. The selections were made primarily on the basis of storage compatibility and product composition. On a preliminary basis, six of these combinations were selected for consideration as propellants for the regeneration period.

Isentropic SPC's were calculated for seven of these combinations to evaluate them for application to internal combustion engines for the non-regenerative period. As expected, hydrogen and oxygen have a substantial advantage over the other combinations. Hydrogen and hydrogen peroxide were second best, and the other five were about equal. A comparison was made between hydrazine and hydrogen peroxide as monopropellants for an internal combustion engine on the basis of isentropic SPC. Hydrazine was found to be superior.

Hydrazine and nitrogen tetroxide are not attractive for the regeneration period, and are not competitive for the non-regenerative period. Therefore, they should be considered as propellants for non-regeneration period engines only if dependable quantities are available from LEM vehicles.

Among the propellant combinations being considered for regeneration are the elements N, H, O, and C. A study was made of possible methods of recovering H_2O, CO_2, CO, and N_2, and subsequently regenerating specific fuels and oxidizers. It was concluded that recovery of these product components is feasible.

Preliminary process designs were made for the recovery and regeneration of the hydrogen and oxygen and the ammonia and oxygen combinations. Energy required, equipment weight, and heat rejection were calculated for a range of capacity. Both of these combinations were found to have a ratio of heat of formation to energy of regeneration of about 0.4

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The isentropic SPC for hydrogen as a non-reactive propellant in an engine such as the cryhocycle were calculated.

The value of solar energy constant for the lunar surface to be used in the study is 129.3 watts/ft^2.

The properties of eight isotope materials which are possible heat sources for lunar engines were compiled.

The November effort will be concentrated on the selection of a "preferred" chemical fuel system. This is essential to provide the lead time necessary for energy converter selection and regeneration plant conceptual design.

II. SYSTEMS INTEGRATION

This section of the October monthly report will describe the progress in the areas of:

Power Uses
Environmental Criteria
Reliability for System Comparison
Comparison Techniques

· The objective of the first three areas is to evolve engineering criteria against which engine and fuel systems will be compared. These engineering criteria are in some cases self evident by reason of specific requirements in the letter request, or they can be derived by an examination of the following general areas.

Environment
Interaction with other base systems (Interface)
Operation
Performance

The comparison criteria will be detailed and in some cases revised as this effort is conducted. For this reason, the comparison criteria have been restated in this report. Integration of all criteria will be accomplished in November, and a final statement will be presented.

A. POWER USES

On October 31, discussions were held with representatives of the Boeing Company's Lunar Base Study Program. The objective of these discussions was to obtain information on the requirements imposed on the engine-fuel system caused by the base evolution mode and activity level. The base evolution mode was distinctly different from that assumed by Westinghouse,

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although the total activity as reflected in power capacity (installed kw) was essentially the same.

BASE EVOLUTION MODE

The scope of application for engines in this system was assumed to include furnishing power for "vehicles and equipment operating independently of the main base" with the output of such independent equipment assumed to be 10 kw initially and increasing to 200 kw in two years. Whereas Westinghouse had assumed that under this definition the number of such engines would range from about 18 (not including requirements under 1/2 kw) in the first year to about 38 in the fifth year, the assumption of Boeing shows 2 engines in the three-month base which would increase to about seven engines in the most extensive base (base 4).

In comparing total installed capacity, a good agreement is obtained. The Boeing approach however assumes that most loads will be satisfied by central station power, and that most activity would therefore be centrally located at the base.

MAINSTREAM APPROACH

The power use section of the September monthly progress report contained a table (page 3-9) which listed the possible uses to which an engine could be put. Included are several uses which would require small amounts of power. Communication relays, landing beacons and some scientific experimentation are examples of small power demands. These individual uses will not be a significant part of the overall base power demand and usage. The letter request contains a ground rule that engine power capacity will range upwards from 1/2 kw. For this reason small power loads such as noted above are not considered to be "mainstream", and the selection of engines will not consider them. Power devices required as an auxiliary on larger

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engines, such as a small battery for fuel cell or dynamic engine startup, will however be included.

The small power loads can best be satisfied by engines specifically designed for the application. A small isotope thermoelectric system furnishing power for a lunar surface communication relay is an example.

LOAD PROFILE MODEL JUSTIFICATION

Some questions have been raised concerning the rational used in selecting the motive, stationary, and portable model loads presented in the first monthly report and utilized in the assumption of an overall base load profile. The following discussion will clarify further the rational used in the model selection.

Table 2-1 presents a compilation of power requirements for a variety of proposed lunar roving vehicles reviewed during a brief literature search. This table indicates the wide variety of vehicles and missions contemplated by researchers in the field. Vehicle operating times range from 8 to 2400 hrs; average locomotion requirements range from 0.125 to 31.6 kw; and the ratio of average auxiliary loads to average locomotion loads ranges from 0.052 to 1.29.

Although requirements showed little agreement due to the variety of missions contemplated, investigation of the literature did provide some general direction for model formulation. The most common vehicle uses were:

Crew and Equipment Transport
Exploration and Sample Gathering
Surface Modification and Base Construction

Generally, the auxiliary power requirements were specified for life support, environmental control, lighting, communication, monitoring and control equipment, and survival operational modes. Only a few of the papers discussed

TABLE 2-1

LUNAR ROVING VEHICLES
LITERATURE MODELS

	A	B	C	D	E	F	G	H	I
Vehicle operation time (hrs.)	8-24	24	168	168	168	200	200	400-675	2400
Speed (mph.)	10	25	?	?	6	1	5	10	?
Crew Size (No. of Men)	2	3	?	?	?	unmanned	?	3	unmanned
Locomotion Loads									
Maximum kw	3.0	21.0	30.0	4.9	--	0.125	--	31.6	4.9
Av. kw	2.75	11.0	--	2.6	0.48	0.1	7.0	23.0	2.6
Av./Max.	0.92	0.52	--	0.53	--	0.90	--	0.74	0.53
Auxiliary Loads	all	all	all	Comm. Navig.	all	all	all	all	Comm. Navig.
Maximum kw	1.0	--	5.0	1.95	--	0.4	2.7	--	1.95
Average kw	--	3.0	--	1.95	.617	0.3	1.0	0.725	1.95
Other Loads	none	none	none	Life Support	none	none	none	none	Construction
Maximum kw	--	--	--	1.73	--	--	--	--	1.0
Average kw	--	--	--	1.50	--	--	--	--	0.8
Av. Aux./ Av. Loco.	0.36	0.27	?	1.02	1.29	3.0	0.143	0.052	0.44
Peak Aux./ Peak Loco.	0.33	0.14	0.17	0.75	?	3.2	?	?	0.40

Note: The nine power demand models represent values extracted from literature on a variety of lunar roving vehicle designs. The model letters correspond to the reporting reference in each case.

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the auxiliary requirements for remote sample gathering hardware, construc-
tion equipment, or engine warmup and starting equipment.

It should be noted that one reason for the uncertainty in the locomotion re-
quirements is the uncertainty of the nature of the lunar surface, and there-
fore, vehicle performance and speed. Consequently, in choosing models, re-
quirements dictated by pessimistic estimates of the lunar surface and low
speed were favored. Further lunar probes should help clarify this decision
and aid in engine sizing.

Table 2-2 presents the results of the literature search of estimated power
requirements for an "S" module type application. All of the models except
model B represent systems utilizing essentially storage of life support expend-
ables. Model B includes a system with CO_2 regeneration.

The table points out a range of .25-.8 kw/man for the functions listed at the
bottom of the table. Our model for the earth supply period (pages 3-11 and
3-12 of first monthly report) favored the high side of this range 0.75 kw/man,
for two primary reasons: First, habitability and power consumption go hand
in hand. Current life support state-of-the-art deals with system exposures
to the space environment of hours to days for very restricted operations.
Our estimate is that as actual flight EC & LS systems progress toward more
habitable ones, power levels will increase accordingly. Second, to justify the
base, we believe that more information gathering and experimentation will be
required than is currently predicted. Therefore, allocations for communica-
tions and experimentation would be larger.

The power level per man shows an increase (pages 3-14, 3-15, 3-16) to 1.75
kw during the years when nuclear power is available. This is an arbitrary
choice, noting that in life support there is a tradeoff between power require-
ments and material expendables. It is assumed that with nuclear power
available more life support regeneration will be accomplished, thus necessi-
tating a higher kw per man requirement with lower logistically supplied ex-
pendables.

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TABLE 2-2

TYPICAL SHELTER MODULE POWER DEMAND MODELS

Comparison Parameters	A	B	C	D	E	F	G
Number of Crewmen	3	6	2	3	5	2	2
Mission Duration	3 months	6 months	?	?	?	?	?
Average Power (continuous) kw	1.2	4.0	1.6	1.2	2.72	0.87	0.50
Peak Power (demands) kw	2.25	5.6	?	?	?	2.70	0.70
Load Factor (Average/Peak)	0.535	0.715	?	?	?	0.322	0.715
Average kw/man	0.4	0.67	0.8	0.4	0.52	0.44	0.25

Note: These seven power demand models represent the results of a brief literature survey of projected power requirements for the following tentative "S" module subsystem functions: A. Atmosphere Supply and Conditioning, B. Thermal Control, C. Life Support, D. Lighting, E. Communications and Data Management, F. Scientific Experimentation. The model letters correspond to the references from which the values were derived.

COMPARISONS WITH LUNAR BASE CONCEPT STUDY

Preliminary information on installed capacity and usage has been obtained from Boeing. These figures are compared with the Westinghouse assumption in the table below. A modified Westinghouse assumption is also included (New Phase V Model).

POWER DEMAND

Time in Base Evolution (months)	Installed Capacity (Kilowatts)		Energy Consumption (thousand kwh)	
	Boeing	Westinghouse	Boeing	Westinghouse
3	7.0	38.0	8.2	70.0
9	32.0	85.0	39.2	115.0
21	226.0	180.0	808.0	225.0
45	327.0	305.0	3017.0	575.0
60	——	310.4	——	584.4
New Phase V Model		609		1504.0

It can be seen that the assumptions of installed capacity are in reasonable agreement. The usage values (kwh) are not. A prime reason for this is in the base operation mode. Boeing has assumed centrally located activity with an almost total reliance on central station power. The use therefore, would be greater than in a diversified type base such as Westinghouse has assumed.

LUNAR ROVING VEHICLE REFERENCES

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EC & LS SYSTEM REFERENCES

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Mason, J. L. and Burris, W. L., "Advanced Environmental Systems", Garrett Corp., IN: Proceedings Manned Space Flight, St. Louis, 1962.

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B. ENVIRONMENTAL CONSIDERATIONS

Selection of an engine system or regeneration plant for a manned lunar base requires consideration of the lunar environment. The environmental requirements are given in Annex B, C, and D of the latter request. This section of the October monthly report will show the derivation of some comparison criteria from these requirements. The effort is not complete, particularly in the radiation area, where a flux vs. time plot will be furnished in the November report.

The section is divided into three main groupings: meteoroids, radiation, and miscellaneous effects. The last group includes gravity, temperature, pressure, and the lunar topographical features. A generalized treatment of each group is followed by a discussion of the effects on typical components of the engine fuel system. A statement of comparison criteria is included where possible.

STATEMENT OF THE LUNAR ENVIRONMENT

Gravity

The gravitational constant on the lunar surface is estimated as 1/6 that of the earth.

Pressure

The lunar atmosphere is very nearly non-existent. The upper limit is estimated to be less than 1×10^{-13} earth atmospheres.

Thermal Environment

1. Solar constant . . . . 443 BTU/(hr - ft^2)
2. Specific heat of moon surface . . . . 0.2 cal/(cm - $^\circ$C)
3. Surface thermal conductivity . . . 3×10^{-5} to 3×16^{-6} cal/cm - sec - $^\circ$C
4. Underlying solid rock thermal conductivity . . . . 4×10^{-3} cal/cm - sec - $^\circ$C

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5. Lunar albedo 0.124

6. Surface temperature ranges

The data given in Annex B, Lunar Surface characteristics, of the R. F. P. Figure 2 will be assumed.

Meteoroid Environment

The meteoroid environment used will be that discussed in letter request, Annex D. A statement of the specific environment is given in subsection 3, which discusses the shielding problem.

Radiation Environment

The radiation model recommended in the letter request, Annex C, will serve as a basis for this study. Further statements concerning the proposed radiation hazards are included in subsection 4.

Lunar Surface Features and Materials

The discussion of probable lunar surface features and materials presented in the letter request, Annex B, will serve as guidelines for the purposes of this study.

TYPICAL COMPONENTS

In general, a multipurpose engine and fuel system can be broken down into the following subsystem groupings with associated components:

Fuel and Exhaust Product Storage and Handling Subsystem

1. Tankage - shielding - insulation
2. Re-fueling equipment
3. Storage conditioning equipment - radiators, etc.

Basic Energy Conversion Subsystem

1. Combustors, boilers, and condensers
2. Heat exchangers
3. Static thermal, chemical, and photovoltaic power generators

4. Prime movers (turbines, reciprocators)

5. Motors, pumps, working fluids, and plumbing etc.

Waste Heat Rejection or Storage Subsystem

1. Radiators

2. Thermal storage reservoirs etc.

Power Conditioning and Control Subsystem

1. Power converters - gear reducers etc.

2. Power conditioning hardware

3. Load switching equipment

4. Monitoring and control equipment etc.

Startup, Shutdown and Maintenance Subsystem

1. Auxiliary energy supplies (batteries, etc.)

2. Maintenance tools and spare parts

METEOROID ENVIRONMENT

One of the primary hazards to men and equipment in the lunar environment is that which arises from the meteoroid bombardment of the lunar surface. Many engine and fuel system components will be vulnerable to this hazard and thus will require some measure of protection or shielding. Radiators, solar collector and concentrator surfaces must of necessity be directly exposed to the environment. Engines and fuel tanks for vehicles or portable power supplies will also be vulnerable. For example, the lunar landing vehicle bus skin, during the required one-year storage period, will represent a large surface area susceptible to meteoroid puncture.

As previously indicated the NASA "Meteoroid Environment of Project Apollo" of Jan. 31, 1963 (given in Annex D of the letter request) will serve to guide equipment designers. As pointed out on page 11 of this annex, the meteoroid hazard on the lunar surface is intensified over that for space because of the

hypothesized secondary particles (ejecta) arising from primary meteoroid impacts. Therefore, in comparing systems, meteoroid shielding weight penalties will be based on penetration probabilities twice those anticipated in space.

A nomograph (Figure 2-1) has been prepared to aid shielding designers. Briefly, this nomograph is based on the following assumptions, which are in turn based on the data given in Annex D of the letter request:

1. The shield penetration phenomenon is defined by the Ames (Charters and Locke) criterion with an uncertainty factor of 5 included to allow for any variations from this criterion.

2. The meteoroid spectrum is limited to spherical particles having a density of 0.5 gm/cc and a velocity of 30 km/sec.

3. A meteoroid will penetrate a thin wall having a thickness 1.5 times the radius of the hemispherical hypervelocity crater.

4. The meteoroid flux is given by the equation

$$N = 4.4 \times 10^{-14 + 0.4 \ M}$$

in which M is the visual astronomical magnitude of a point source and N is the cumulative flux of meteors brighter than magnitude M.

5. The initial kinetic energy of a meteor is defined by the equation (in MKS units)

$$mv^2 = 3.7 \ Lv^{0.3} \times 10^{4 - 0.4 \ M}$$

in which L is a dimensionless constant assigned a value of 2.78 to make the zero magnitude 30 km/sec. meteoroid weigh 2.5 grams.

The physical characteristic factor, K, for the various possible shield materials is plotted in Figure 2-2 as a function of the material temperature.

To show the importance of the various parameters pertinent to meteoroid shielding design, typical material thicknesses for a particular set of conditions
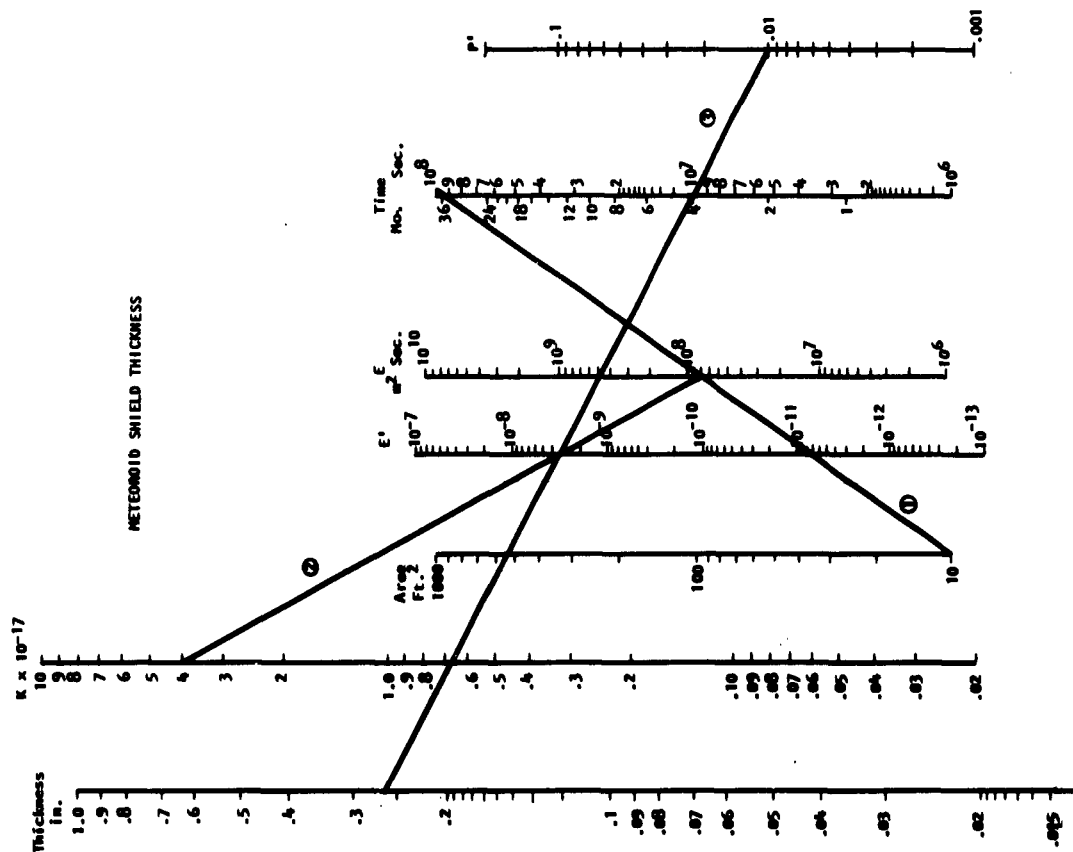
METEOROID SHIELD THICKNESS

Figure 2-1

Figure 2-2. Penetration Susceptibility Factor for Prospective Shield Metals

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are compared for various exposure times in Figures 2-3, 2-4, and 2-5. It is readily apparent from these curves that a substantial penalty must be paid to obtain a high probability of no punctures. The skin thickness required for a 90% probability must be increased 2 to 3 times to realize a 99% probability of no punctures, or 4.5 to 5 times to increase this probability to 99.9%. This consideration strongly suggests the desirability of dividing those system components especially susceptible to meteoroid damage, such as radiators, into segmented units each capable of assuming only a fraction of the total heat load.

The advantages of segmented radiators from a reliability and weight standpoint for space applications has long been recognized. To arrive at a better appreciation of the relative advantages of segmented radiators, a short statistical study was performed, the results of which are shown in Figures 2-6, 2-7, 2-8, and 2-9. In these curves, an over-all radiator has been assumed with a combination of protective armor, exposure time and meteoroid flux characteristics to give several values of over-all radiator probability of survival--which is defined as the probability that the particular radiator over the period in question will not receive one puncture hit. Each set of curves is based on a different value of this probability which we have taken as 2%, 30%, 70% and 90%. If this radiator is then cut into a number of independent segments (and we have chosen 2, 5 and 10 in this analysis) we find still the same probability of a single puncture. However, the probability of multiple punctures or the probability of more than one segment being punctured is very much lower. Therefore, the probability of survival versus the percentage of the radiator that will survive has been plotted. A brief examination of these curves indicates the tremendous advantage of radiator segmentation. As an example of this process taken to a logical extreme, a 100-segment radiator which before segmentation has a 0.000026 chance of survival has a 0.9999+ chance of being 77% intact. (See Figure 2-10.)
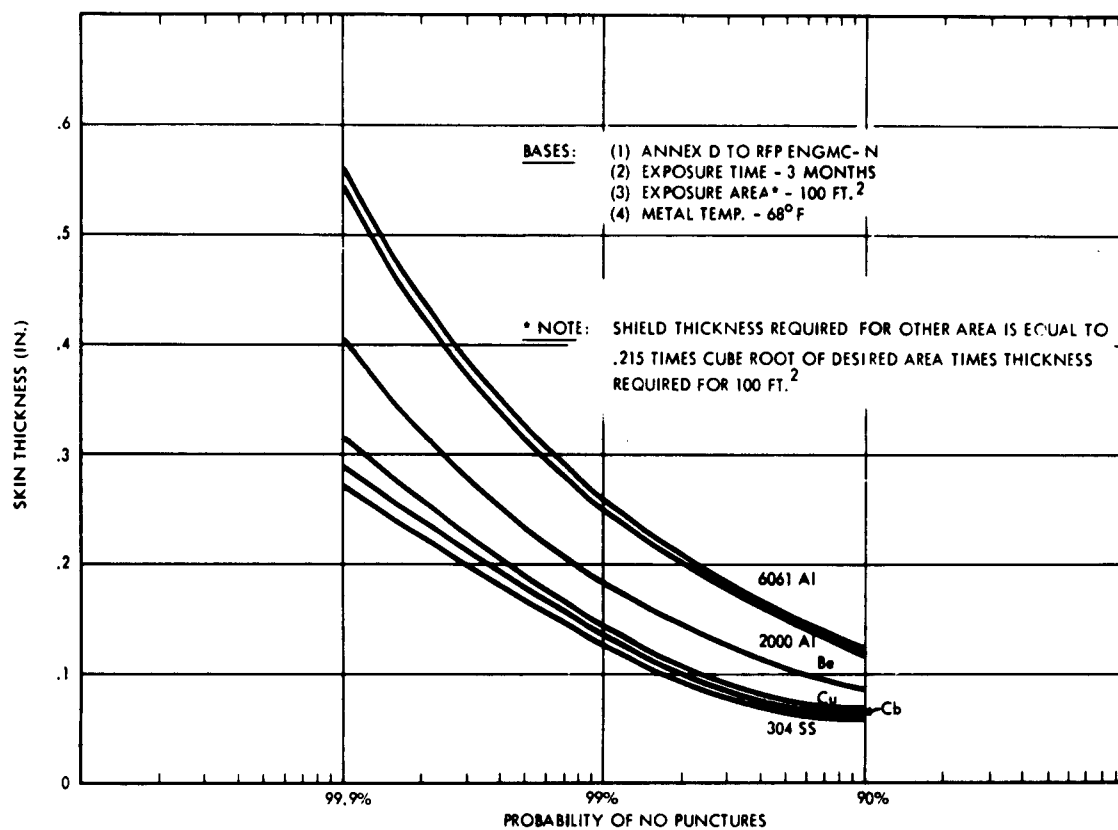
Figure 2-3. Meteoroid Shield Thickness Requirements on the Lunar Surface

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BASES: (1) ANNEX D TO RFP ENGMC - N
(2) EXPOSURE TIME - ONE YEAR
(3) EXPOSURE AREA* - 100 FT.
(4) METAL TEMP. - 69° F

*NOTE: SHIELD THICKNESS REQUIRED FOR OTHER AREA IS EQUAL TO .215 × CUBE ROOT OF DESIRED AREA TIMES THICKNESS REQUIRED FOR 100 FT².

6061 Al
2000 Al
Be
Cu
Cb
304SS

SKIN THICKNESS (IN.)

PROBABILITY OF NO PUNCTURES
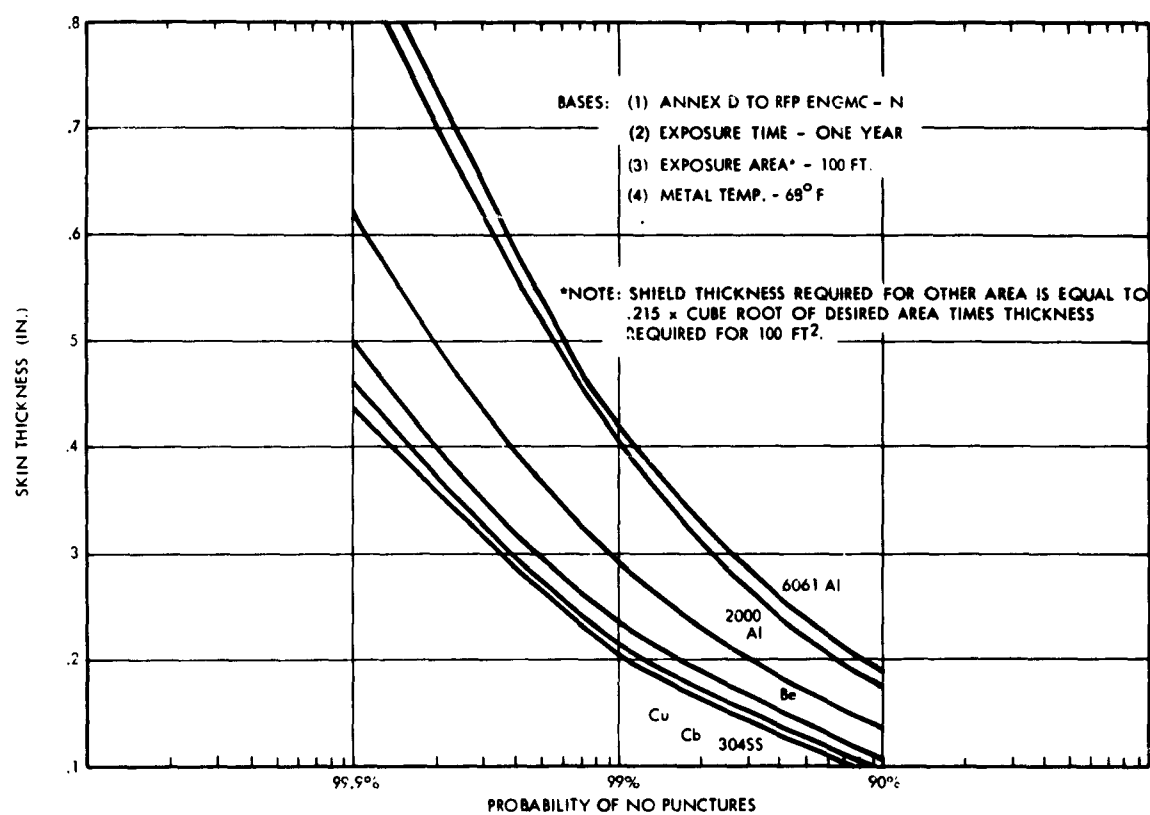
Figure 2-4.   Meteoroid Shield Thickness Requirements on the Lunar Surface

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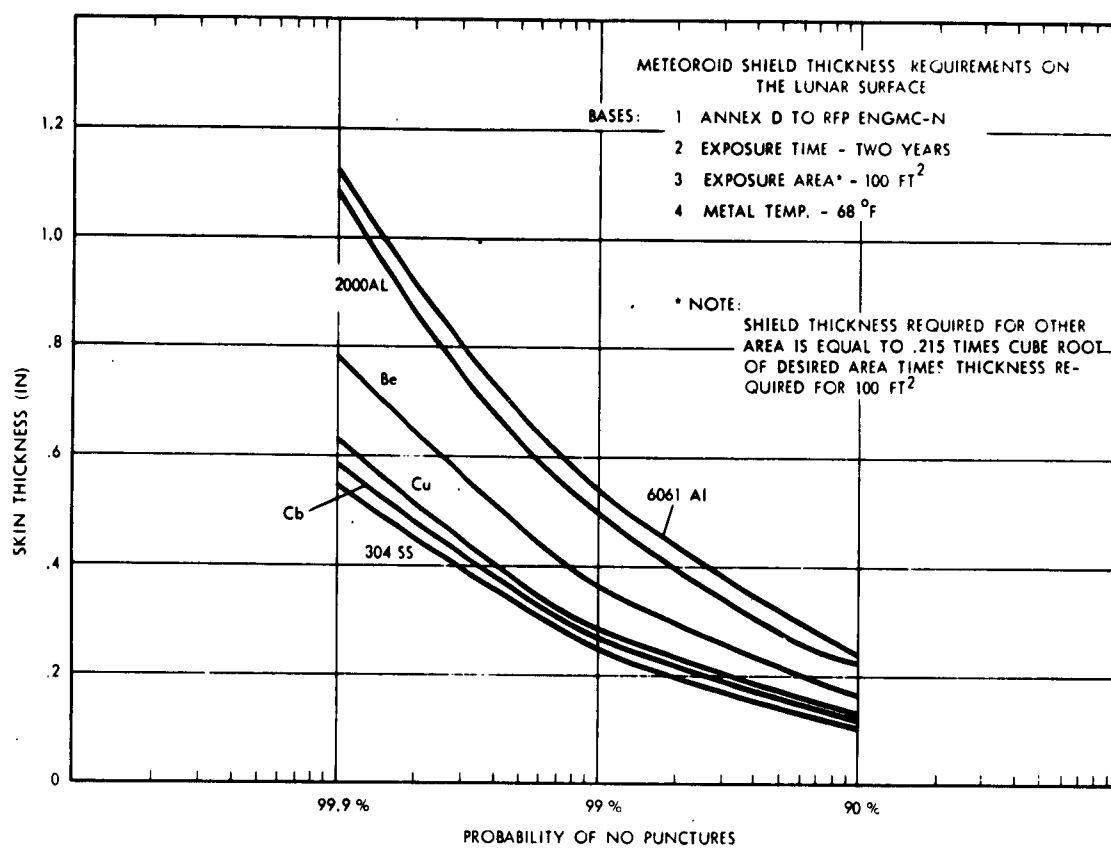
Figure 2-5. Meteoroid Shield Thickness Requirements on the Lunar Surface

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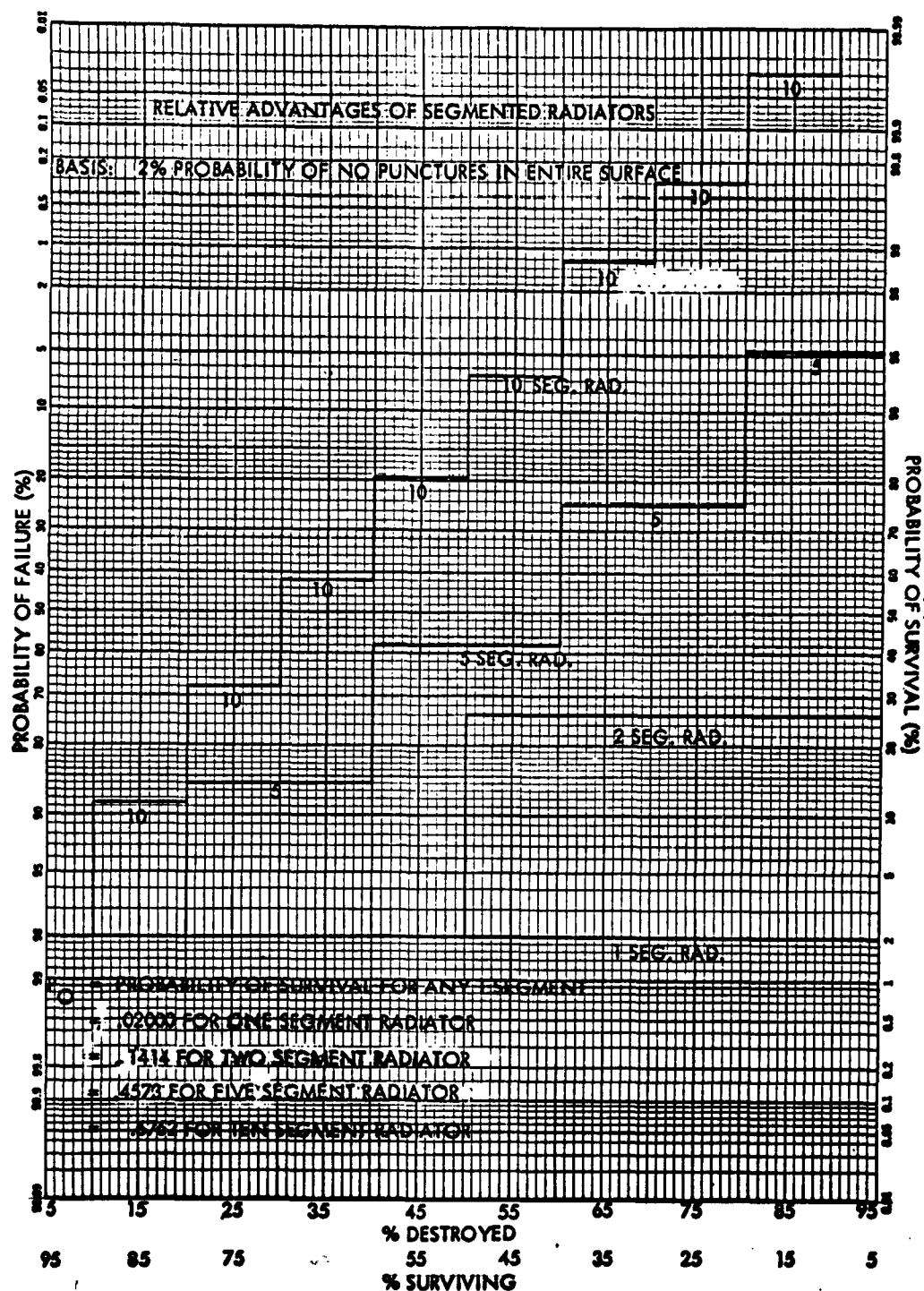
Figure 2-6.   Relative Advantages of Segmented Radiators
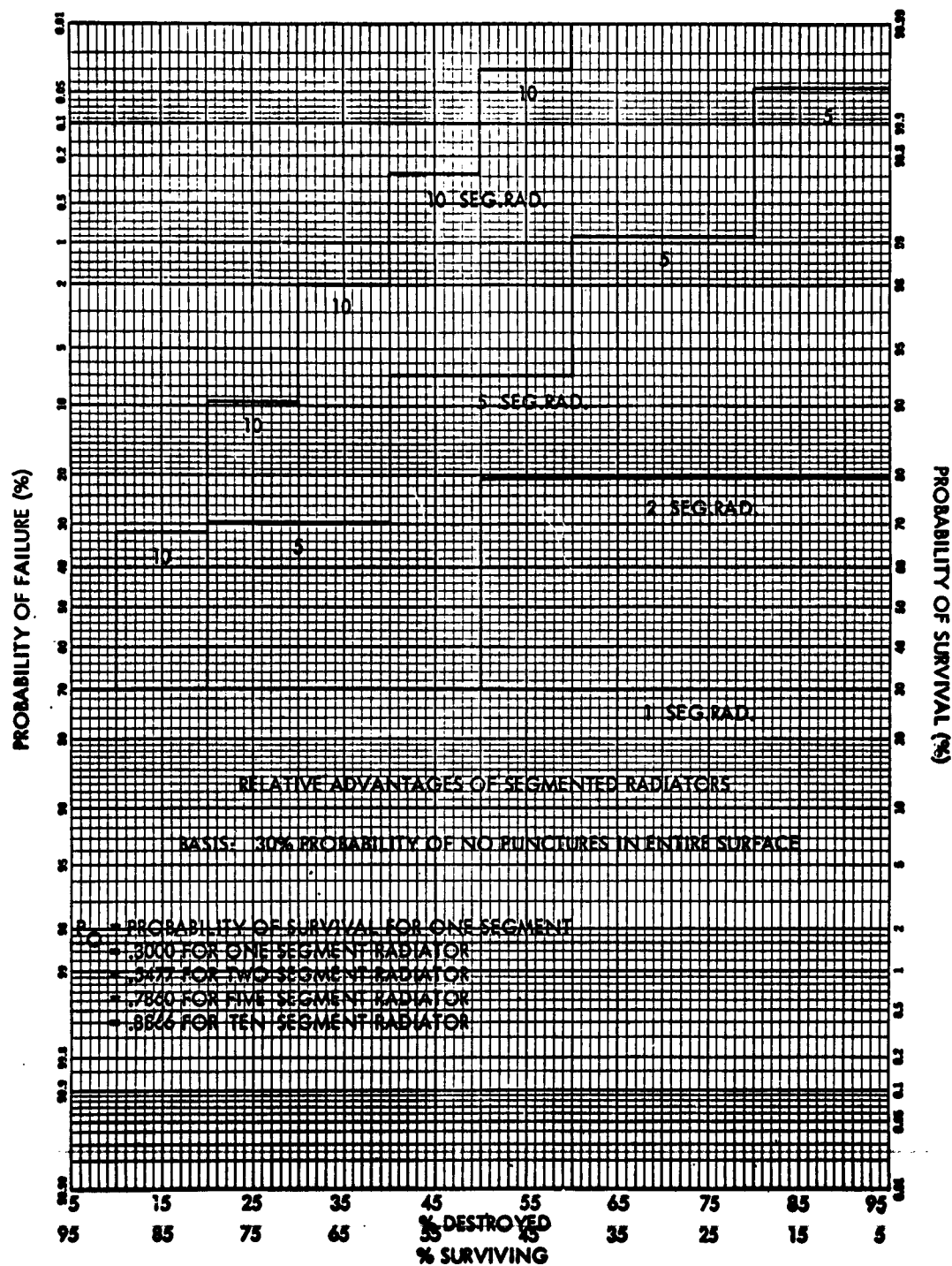
Figure 2-7. Relative Advantages of Segmented Radiators

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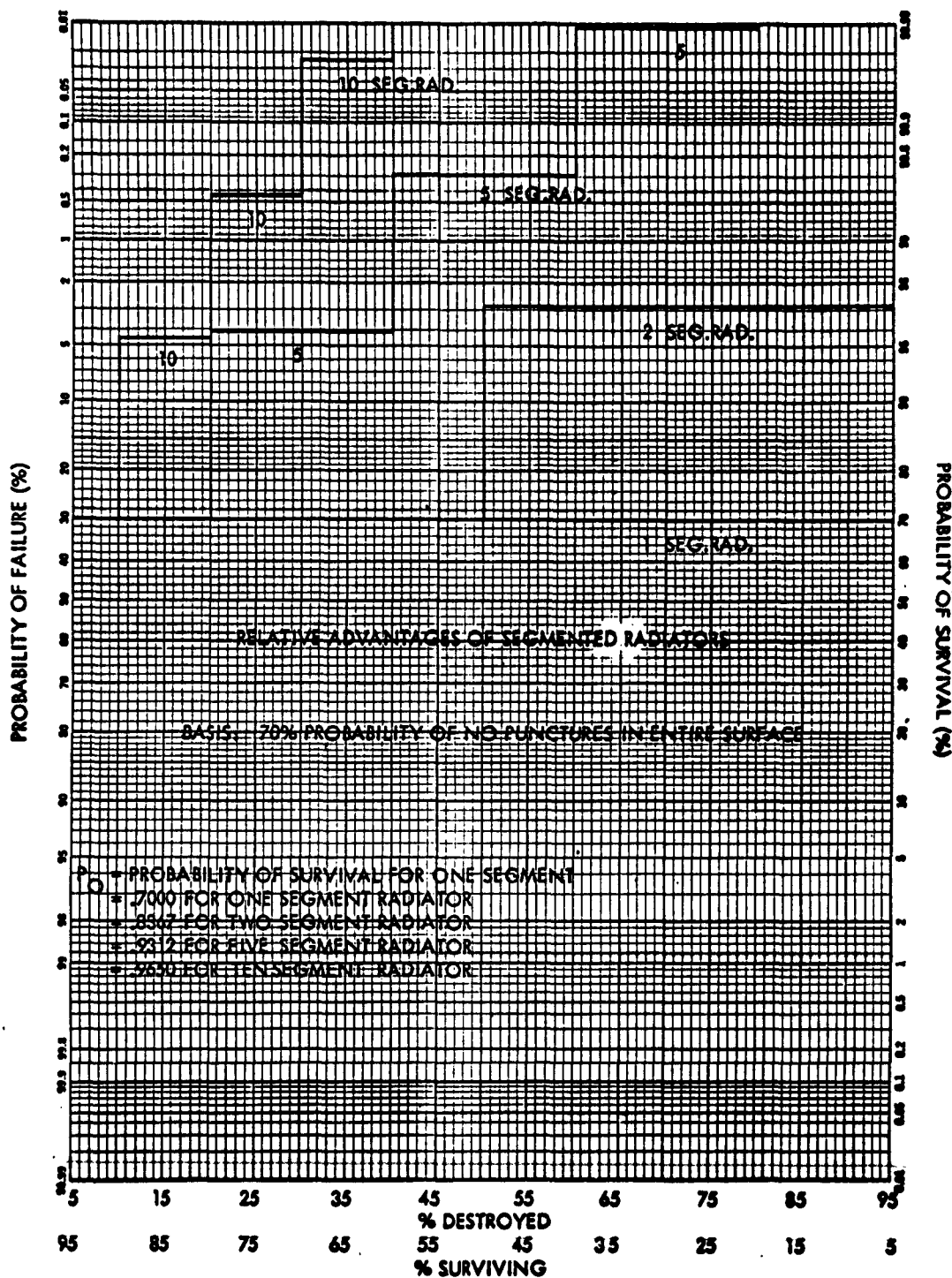
Figure 2-8. Relative Advantages of Segmented Radiators

2-21
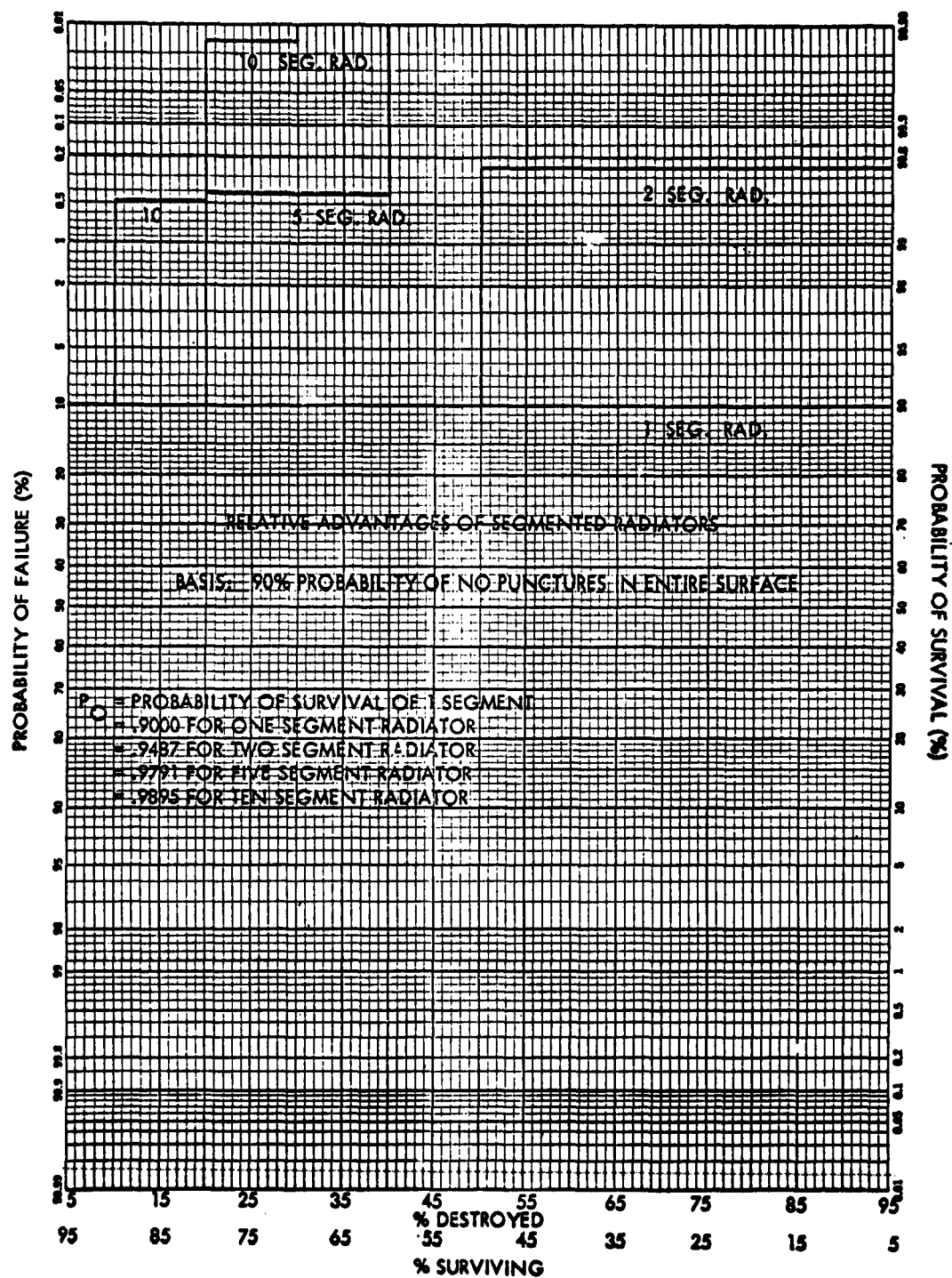
Figure 2-9. Relative Advantages of Segmented Radiators
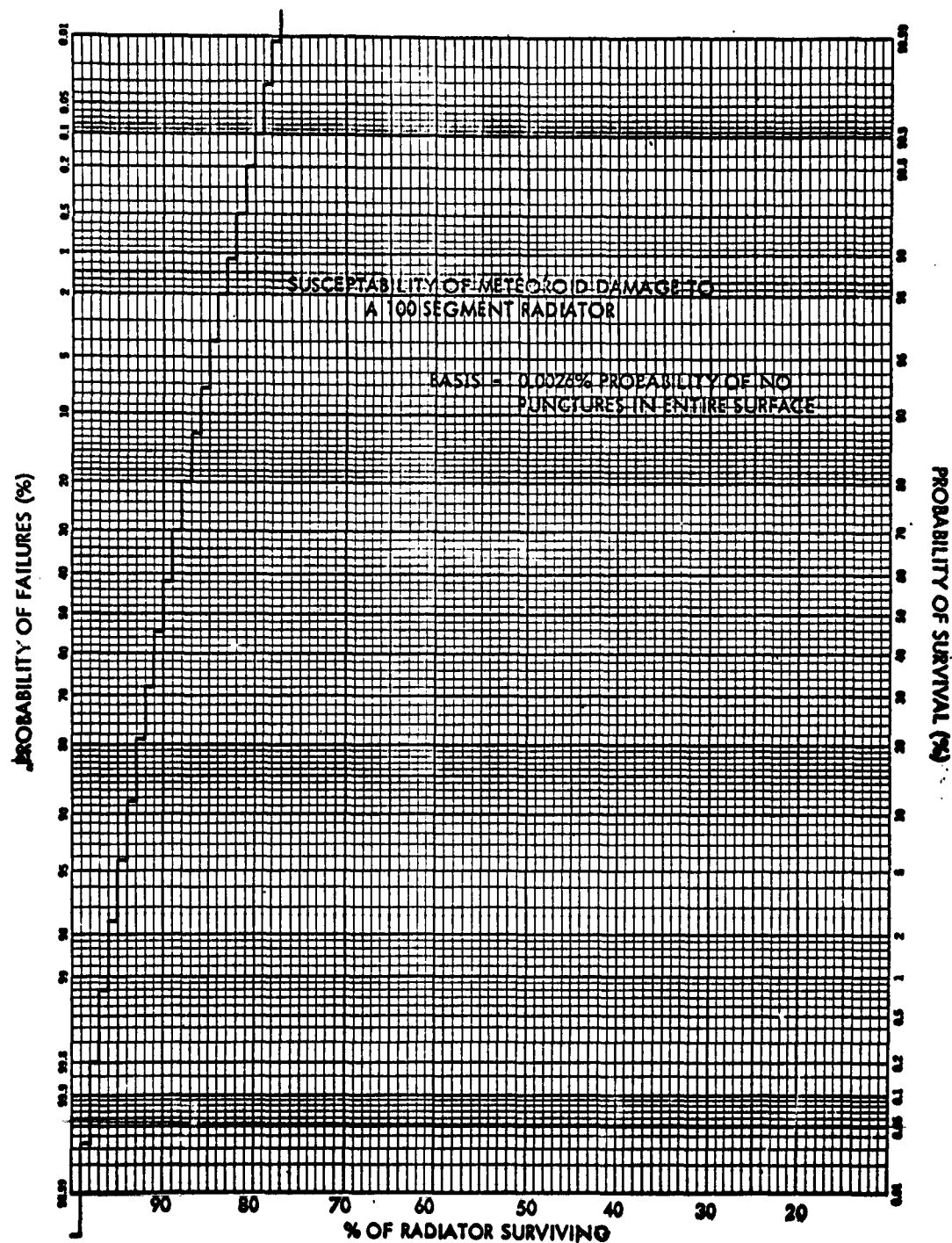
Figure 2-10. Susceptibility of Meteoroid Damage to a 100-Segment Radiator

2-23

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RADIATION ENVIRONMENT

Radiation in space can be traced to three main sources:

1. The Van Allen radiation belt, an intense radiation zone of magnetically trapped particles surrounding the Earth.

2. Galactic cosmic radiation forming a high-energy, low-flux background throughout space.

3. Solar radiation characterized by outbursts of high intensity proton radiation known as solar flare.

In this section, characteristic radiation from each of these sources is examined to provide an understanding of its formation, energy distribution, particle flux, and methods of propagation.

Van Allen Radiation

The existence of a high intensity radiation zone surrounding the earth was confirmed in 1958 by the Explorer I and III Satellites. This zone, since found to consist of geomagnetically trapped particles, is known as the Van Allen Radiation Belt or "Magnetosphere".

Charged particles enter the magnetosphere from two basic sources. The first source is the sun, producing high energy electrons and protons which, during a solar flare event, are injected into the outer portion of the Van Allen Belt. Cosmic radiation is the second source of particles. High-energy cosmic radiation reacts with the nuclei of atmospheric oxygen and nitrogen producing neutrons with a half-life of approximately 12 minutes. The neutrons thus formed decay to produce protons, electrons, and antineutrinos. The antineutrino, a form of energy with charge and rest mass equal to zero, is not affected by the geomagnetic field and quickly escapes. The electrons and protons, however, become captured in the inner portion of the radiation belt.

The path of a charged particle (electron or proton) in the Earth's magnetic field is the combination of three principal motions. The trapped particle

travels with a helical motion along a line of geomagnetic force. While executing this first motion, the particle penetrates the magnetic field approaching the polar latitudes until it reaches a point of sufficient field strength to reverse its direction. It then travels in this new direction until reaching the equivalent latitude at which time it again reverses. In addition to this spiraling between two latitudes (mirror points), the particles exhibit a slow longitudinal drift, positively charged particles drifting westward, and negatively charged particles eastward. This motion and the resulting envelope would appear as shown in Figure 2-11.

Since particle motion in the magnetosphere is not random, but rather is dependent upon the field lines surrounding the Earth, the intensity of radiation measured by space probes is highly dependent upon the position, time, and direction of observation. This fact, combined with differing threshold values of the radiation counters used, has caused great variation and even direct contradiction regarding the particle flux and energy distribution in the magnetosphere. Due to the reasons given above, early data was interpreted as indicating an inner and outer radiation belt. Latest data transmitted by the Explorer 14 Satellite indicates that the magnetosphere is a single large trapping region in which particle characteristics vary greatly.

The latest available information concerning the energy and particle flux in the radiation zone is given in Table 2-3.

Galactic Cosmic Radiation

Galactic cosmic radiation consists of charged atomic nuclei which have been slowly accelerated elsewhere in our galaxy or beyond to very high energies by a means involving charged particle collisions with moving magnetic fields in space. The energy of these particles ranges from 10^8 to 10^{19} electron volts.

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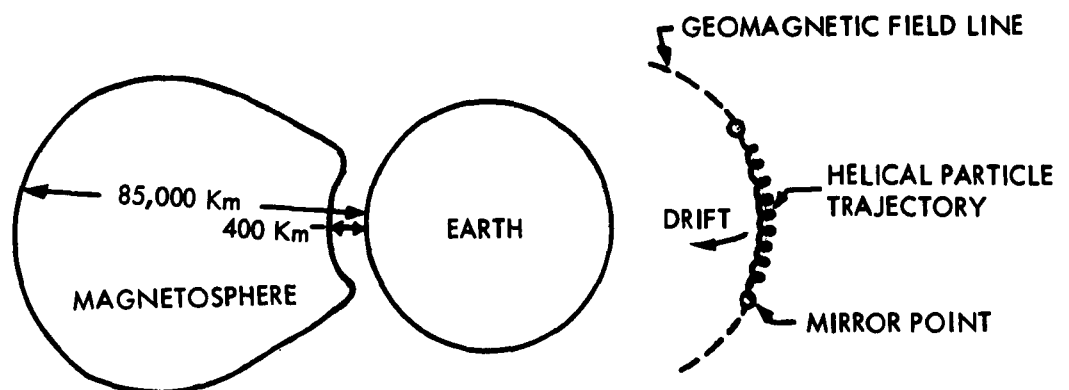
Figure 2-11.   Particle Motion and Envelope of Van Allen Belt

TABLE 2-3

PARTICLE FLUX AND ENERGY IN THE VAN ALLEN BELT

Altitude Above Earth, In Km	Particle Energy	Flux Particles/cm^2 - sec
3,600	electrons > 20 Kev	2×10^9
	electrons >600 Kev	10^7
	protons > 40 Mev	2×10^4
10,000	electrons > 20 Kev	10^6
	protons > 40 Mev	3×10^4
20,000	electrons > 20 Kev	10^{11}
	electrons > 1.5 Mev	10^4
	protons > 30 Mev	10^0
31,000	electrons > 40 Kev	10^8
	electrons >230 Kev	5×10^6
	electrons > 1.6 Mev	1×10^6
39,000	electrons > 40 Kev	1.5×10^8
	electrons >230 Kev	1.5×10^6
	electrons > 1.6 Mev	2×10^5
55,000	electrons > 40 Kev	2×10^7
	protons >500 Kev	5×10^4
	electrons >230 Kev	3×10^4
	electrons > 1.6 Mev	1.5×10^3
65,000	electrons > 40 Kev	1×10^6
	protons >500 Kev	1.5×10^4
	electrons >230 Kev	6×10^3
	electrons > 1.6 Mev	2×10^2
80,000	electrons > 40 Kev	5×10^2
	protons >500 Kev	5×10^2

Abundance	Type of Particle
86.5%	Hydrogen nuclei (protons)
12.4%	Helium nuclei (α - particles)
0.2%	Light nuclei (Li, Be, B)
0.7%	Medium nuclei (C, N, O, F)
0.2%	Heavy nuclei

Near the Earth's orbit, this cosmic ray flux is modulated by the eleven-year solar cycle. Galactic radiation is deflected by the magnetic fields associated with solar radiation resulting in an inverse relationship between solar activity and cosmic radiation intensity. As the solar activity increases to its maximum, the cosmic intensity decreases by a factor of two, and the particle flux decreases from 3 particles/cm^2 - sec. to 1 particle/cm^2 - sec. The average kinetic energy of galactic cosmic radiation is 3.6 - 4.0 Bev resulting in an integrated radiation dose at solar maximum of .5 rem/week.

Solar Radiation

The primary source of penetrating radiation is the sun. Solar radiation, as produced by fusion reactions, consists basically of protons and α - particles but may also contain electrons, neutrons, X and gamma ray photons, and heavy nuclei.

Solar radiation is composed of a steady output of charged particles with energies less than 3 Kev, and solar flare events which produce particles possessing much higher energies. Although the average energy of a solar flare particle is less than that of galactic cosmic radiation, the very large number density of a solar event causes it to be of greatest importance when considering the radiation hazards of a space environment.

Solar flares or "storms" originate when explosions on the sun cause clouds of magnetized gas to be expelled from the sun at velocities up to 700 miles/ sec. As the expanding gas cloud envelops the Earth, a magnetic coupling takes place allowing high-energy nuclei from the sun to travel in a helical trajectory along the magnetic lines of force with relativistic speed. Depending upon their heliographic origin, these particles reach the Earth's orbit

one to eighteen hours after the occurrence of a flare. Since, as previously stated, the propagation of solar particles is related to the magnetic coupling of the Sun and the Earth, travel time is highly dependent upon the magnetic conditions existing at the time of the flare and thus upon the previous solar activity.

The solar storm may be classified as either relativistic or non-relativistic. The occurrence of relativistic events shows no definite correlation with the 11-year solar cycle, the events occurring randomly at a rate of less than one event per year. Non-relativistic events, however, occur principally during the three years following the sunspot maximum and occur at the rate of one per month.

The predictability of solar flare events is a topic of much controversy. It has been observed that at the onset of a flare event, increased ionization occurs in the atmosphere above the magnetic polar region. This effects an increase in the absorption of galactic radio noise, an occurrence known as a polar cap absorption event (PCA). The particle flux and energy spectrum of a solar event can thus be determined by observing the PCA with ground based radar. This type of observation would at best produce only short-range warning of an event.

Since flares always occur in the vicinity of large sunspot groups, it is possible that future developments in the observation of sunspot phenomena will permit forecasting of large flares for periods of up to several days. Observation of PCA and sunspot events indicate that approximately 90% of the time during solar maximum would be theoretically available for safe space flight.

Radiation Attenuation

Comparison of the present estimates of space radiation with the allowable radiation dosage for man confirms the existence of a major environmental hazard greatly affecting future manned space flight. The shielding required

to reduce the radiation absorbed by astronauts to a safe level will consume a significant portion of the mission's payload. For this reason, the problem of shield optimization to achieve minimum weight and volume has been the subject of extensive research and investigation in recent years.

The mechanism of radiation attenuation in passive bulk material has been studied to evaluate the shield weight required to absorb space radiation or reduce its energy to safe levels. Active systems, relying on the production of electrostatic or magnetic fields to deflect incoming particle radiation, have also been investigated.

This section summarizes the findings concerning optimum shielding materials and systems for the various types of primary and secondary radiation to be encountered beyond our atmosphere.

Protons

As protons penetrate a shield material, collisions with bound electrons cause a linear decrease in proton energy. The energy given up by the proton is absorbed by the stopping material through processes of ionization and excitation of atoms. Energy transferred per gram/cm^2 of shield material is known as the rate of energy loss (REL), and is a function of the original particle energy and the atomic composition of the stopping material. Figure 2-12 gives the REL for protons in various materials. Figure 2-13, a graph of proton range vs. energy for several materials, shows that low E (atomic number) materials are most effective for proton shielding.

The higher energy protons may also interact with the shield nuclei resulting in the formation of secondary radiation. Alpha particles, neutrons, protons, and gamma rays are commonly produced by such a process. Due to its high penetration, the neutron is the most dangerous of the secondaries. As can be seen from Figure 2-14, the production of secondary particles increases with increasing atomic number of the shield material for higher energy incident protons.
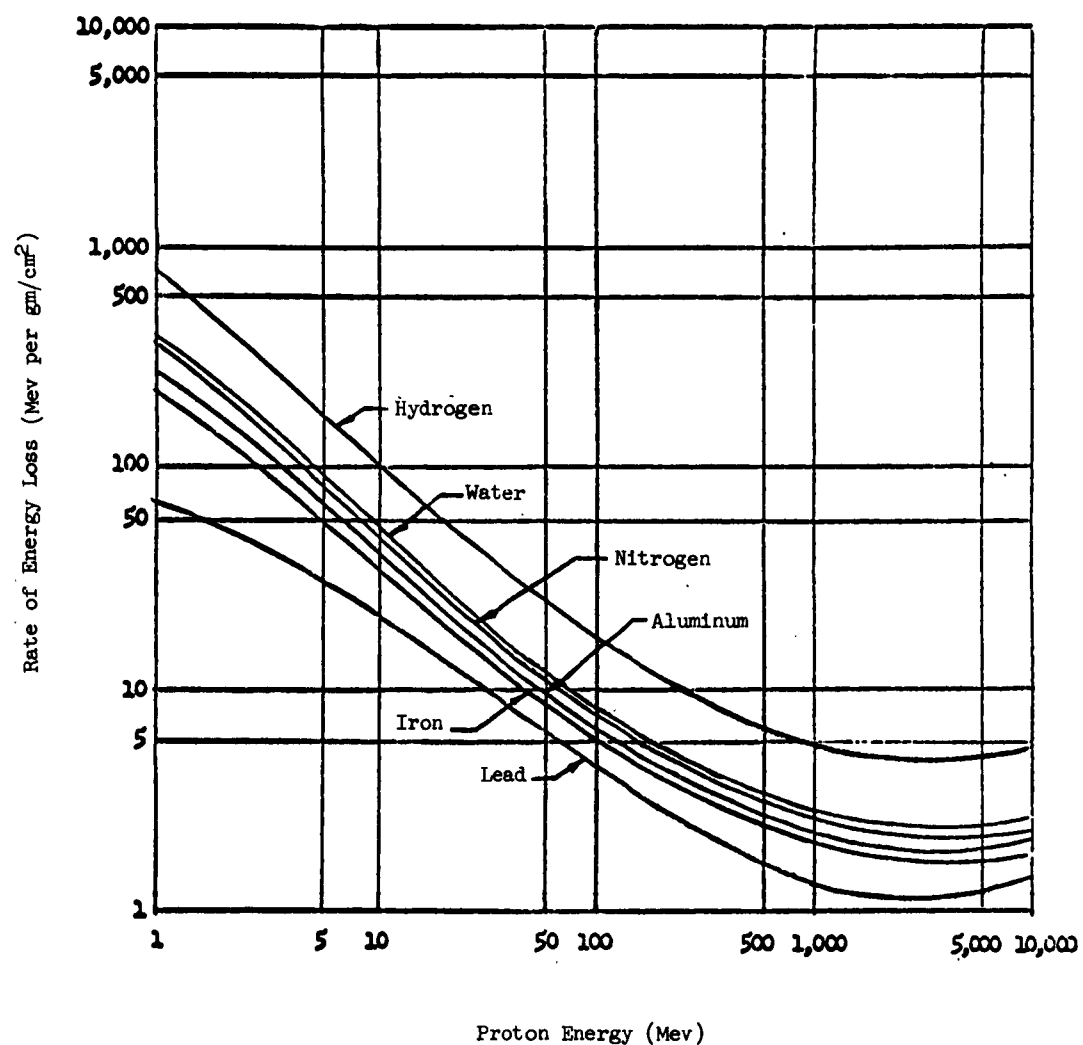
Figure 2-12. Rate of Energy Loss of Protons

Proton Energy (Mev)

2-31
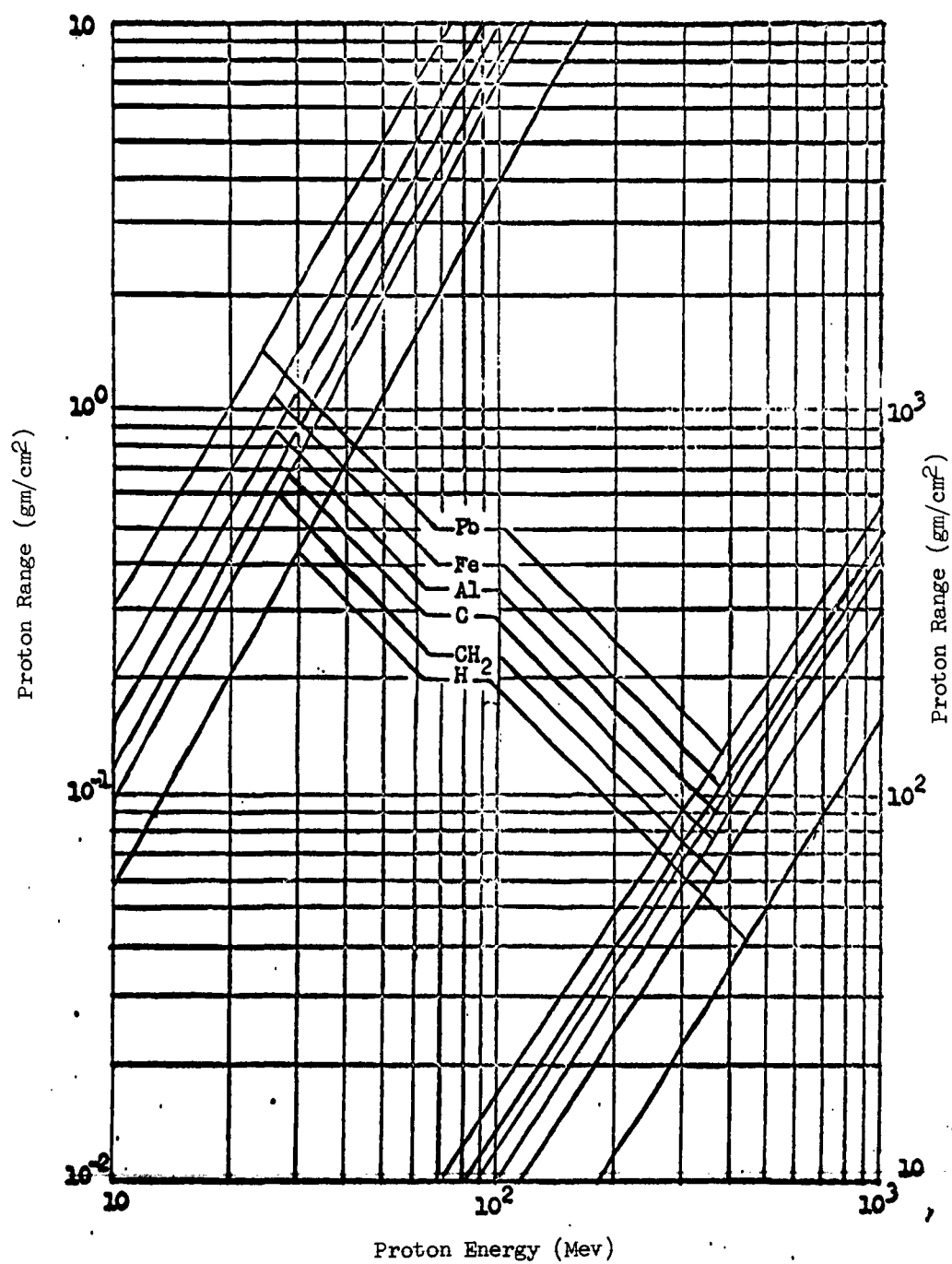
Figure 2-13.   Proton Range Vs. Energy
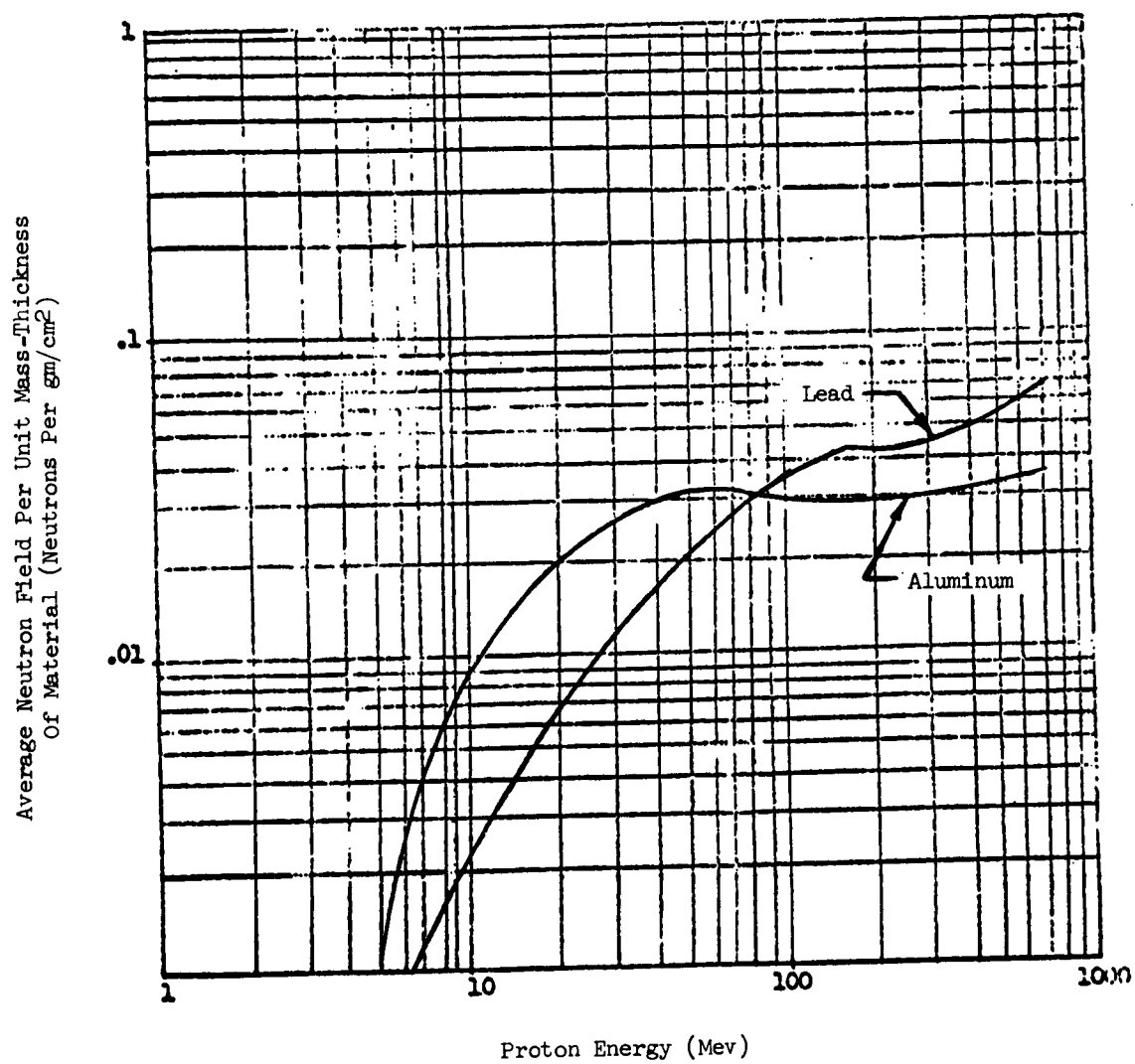
Figure 2-14. Production of Secondary Radiation Vs. Proton Energy

2-33

Electrons

Electrons penetrate a shield material with interactions similar to those of protons. As in the case of the proton, the charged particle collides with the electrons of the shield atoms resulting in their ionization and excitation. The penetration of electrons in aluminum has been plotted in Figure 2-15.

Electrons may also be decelerated by interactions with the electric fields surrounding the shield atoms. This deceleration results in the release of electromagnetic energy (bremsstrahlung) in the form of X-rays. The radiation emitted ranges in energy up to that incident electron and is of importance when considering electron radiation dosage, due to its deep penetrating ability. As shown in Figure 2-16, the production of bremsstrahlung radiation is proportional to the electron energy and to the atomic number of the shield material, indicating that the use of low Z material is desirable for high energy electron shielding.

X and Gamma Rays

This short-wavelength electromagnetic radiation is encountered as secondary radiation from proton and electron interactions in a stopping material. Attenuation occurs by collision with bound electrons, photoelectric absorption, nuclear interaction, and electro-positron pair production. The attenuation due to the combination of these effects can be presented by the mass absorption coefficient, μ, where the fraction of X or γ-ray photons remaining after traversing χ gm/cm^2 of material is given by $e^{-\mu \chi}$. Figure 2-17, showing the mass absorption coefficient for various materials, indicates that lead would be the most desirable shielding material for secondary electromagnetic radiation over a range of energies.

Neutrons

The neutron possesses no charge, and therefore, does not interact with the electric fields of shield atoms. Neutron attenuation is restricted to collisions with the nucleus of the atom. Absorption of the neutron may occur if a nuclear reaction takes place upon collision. Moderation of neutron energy

2-34
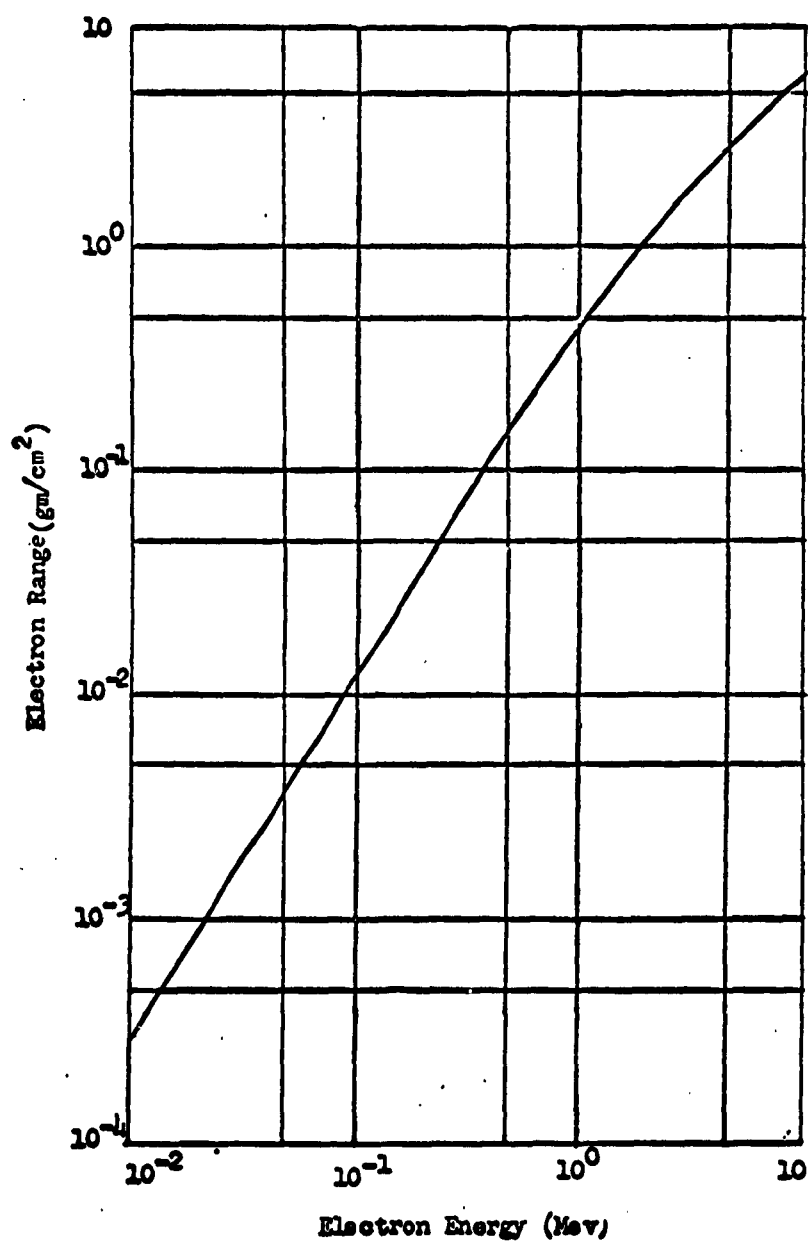
Figure 2-15. Range of Electrons in Aluminum
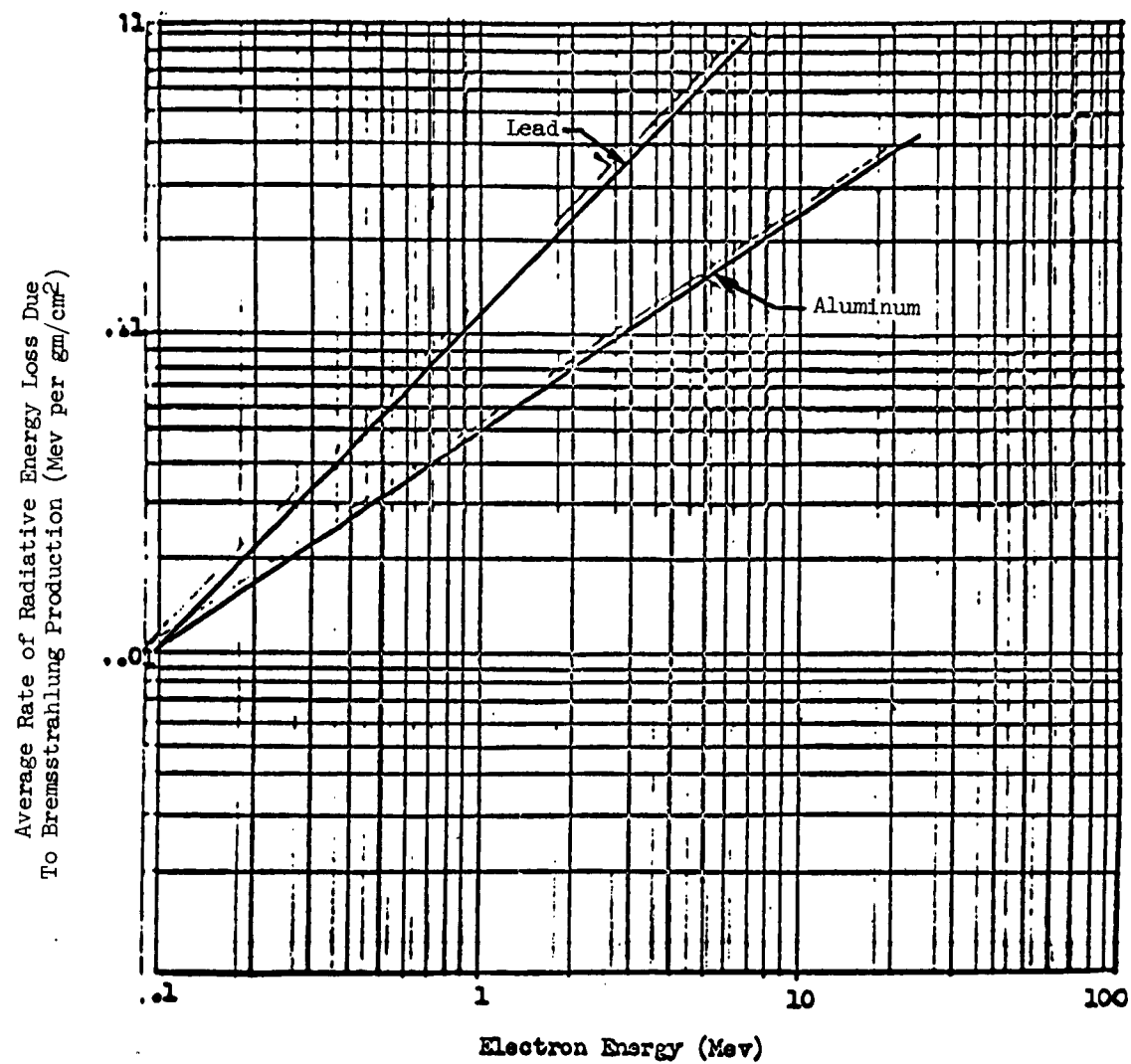
Figure 2-16. Production of Secondary Radiation Vs. Electron Energy
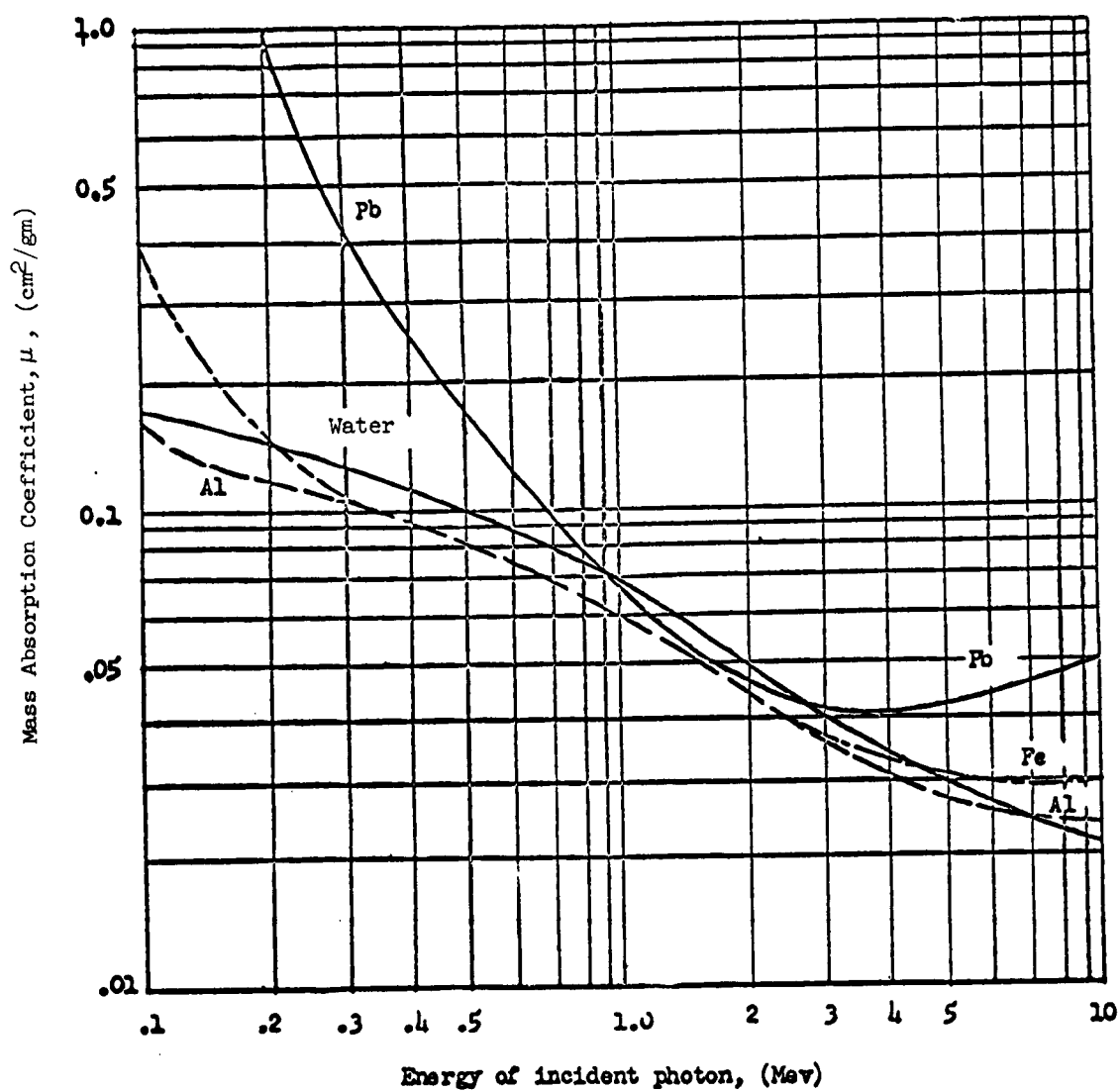
Figure 2-17. Mass Absorption Coefficient of X and γ Rays

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occurs if the collision results in elastic scattering of the neutron. In the latter process, the maximum energy which may be transferred is a function of the mass of the shield nucleus. For a nucleus of mass M, the fraction of the incident energy which may be transferred is given by $M/(M + 1)^2$. Thus, a low Z material would be desirable from the standpoint of reducing neutron energy through the process of moderation. For removal of neutrons by absorption, a material of high neutron cross section such as boron, cadmium, or lithium would be required.

Electrostatic and Electromagnetic Systems

The theory of active shielding for radiation protection is based on the use of electrostatic or magnetic fields of sufficient strength to deflect incoming charged particles and prevent their reaching the shielded area.

In the case of electrostatic shielding, a positive charge would be accumulated on a shell enclosing the space to be shielded. Protons with initial energy less than the potential on the sphere would be repelled by the electrostatic field surrounding the shell. This method does not provide protection against particles with charge opposite to that of the shell (electrons) or against uncharged particles (neutrons). To resolve this deficiency, it has been suggested that the shield be constructed of a positively charged outer shell concentric with a negatively charged inner shell, the charge on the inner shell being approximately one half of that on the outer shell. Such a configuration would theoretically offer protection against both positive and negative particles. The power required to charge the shield varies directly with shelter size and rate of charging. With regard to the latter, a shield maintained at maximum potential throughout the entire mission would obviously require less power than a shield system which must be charged in a relatively short period of time (30 minutes) after warning of a solar event. Some of the problems which must be overcome in the design of electrostatic shelters are: development of light-weight high-voltage generators (approximately 500 million volts); development of light-weight insulating

structures to carry the forces resulting from electrostatic attraction of
charged concentric shells; and the development of methods to reduce field
emmision from such a shelter.

Electromagnetic shielding provides a deflecting force against both positive
and negative particles. To produce the required magnetic field, a coil of
superconducting material would be initially charged and then maintained at
cryogenic temperatures by means of proper insulation and a liquid helium
coolant. Since the protection offered by a magnetic system varies greatly
with the shape of the field, numerous coil configurations have been proposed.
These configurations can be classified as either confined or unconfined field.
The unconfined field extends outward from the shelter with intensity decreas-
ing as the distance cubed, while the confined field is maintained within the
boundaries of the coil. A design utilizing an unconfined field surrounding a
toroidal shelter has been proposed by Levy. The coil for such a shield
system would be made from a hollow tube of superconducting nobium-tin.
Other designers have considered the use of a confined field. Confined fields
eliminate the problems caused by an intense magnetic field extending outward
from the shelter, but have the disadvantage of allowing the incoming particles
to strike the outer coil structure resulting in increased production of sec-
ondary radiation. Magnetic shielding appears to have a weight advantage
over bulk materials for applications requiring the protection of large volumes
against very high energy radiation, a condition which would occur during
prolonged interplanetary flight. The problems involved in the practical use
of such systems include fabrication of superconducting coils, design of cool-
ant and insulation schemes to maintain cryogenic temperatures, and problems
concerning communications in the area enclosed by the field.

Summary
To summarize the attenuation of various types of radiation, the following
chart is given:

Type of Particle or Radiation	Applicable Spacecraft Shielding
Protons	Carbon, hydrocarbons, water, or light metals. Electrostatic and electromagnetic shielding also suitable for higher energies.
Electrons	Same as protons except for the addition of a thin inner shield of lead to stop secondary X-radiation.
X & Gamma Rays	Lead or other high Z metal.
Neutrons	Water, hydrocarbon, or any low Z material for moderation. Boron, cadmium, or lithium for absorption.

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MISCELLANEOUS ENVIRONMENTAL EFFECTS

The following discussion presents some of the remaining environmental effects on possible engine and fuel system processes and equipment.

Reduced Gravitational Constant

From earth departure to installation on the lunar surface, the lunar base engine and fuel system will experience a range of effective gravitational accelerations. Although systems will be designed to withstand the anticipated liftoff and landing forces (8 and 15 g. respectively) while in a non-operational state, they must also be designed to operate in the reduced gravitational field on the lunar surface (approximately 1/6 earth gravity).

To define the extent of the reduced gravity effects in lunar base Engine and Fuel systems, the physical processes and components sensitive to body force considerations must be listed. One logical grouping is:

Fluid Mechanics

1. Hydrostatics: reduced pump NPSH.
2. Hydrodynamics: flow of two phase mixtures in pipes, pumps, heat exchangers, and radiators.
3. Hydrodynamic Lubrication: fluid lubricants in bearings.

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Mass Heat Transfer

1. Free convection in closed systems.
2. Pool boiling and condensation processes.

Those systems using fluids will require stable and predictable pressure drop and heat transfer characteristics. Pumps designed for liquids become inefficient and cavitate if gas or vapor is present at the pump from a slugging condenser. Flow instabilities in heat exchangers and poor contact of the working fluid with the heat transfer surfaces of heat exchangers could cause overheating and surface melting. Turbines designed for use with vapor suffer structural degradation due to the introduction of liquid masses from boiler carryover. Thus, complete conversion of liquid to vapor or vice-versa is an essential design consideration for boilers and condensers.

Temperature

System designers will face the problems associated with temperature variations as great as $500^\circ F$ on the lunar surface. Due to these variations, problems associated with changes in material properties, differential expansion of materials, and sublimation and evaporation will require consideration.

Much experimental work is currently being conducted on material properties at cryogenic temperatures. Such work has revealed some interesting results concerning strength and ductility changes for metals and alloys. While yield strength values tend to increase over room temperature values, brittleness becomes a problem. For cryogenic propellants, brittle weld failure in aluminum tankage have been reported. Such an effect could warrant use of a glass-filament-wound tanks.

The problem of mating surfaces with identical coefficients of thermal expansion is critical, especially for:

1. Vacuum seals
2. Selective coatings on radiator surfaces
3. Fueling connections and by-product handling equipment disconnects

Leakage at vacuum seals on fuel tanks results in loss of fuel, while cracking and peeling of selective coating resulting from thermal cycling can seriously alter the cold sink temperature of a radiator, thus degrading the efficiency of the energy conversion device.

Differential expansion in fuel-handling equipment shocked by cryogenic fuels temperatures could lead to stress concentrations, failure of seals, and eventual fuel leakage.

Finally, sublimation and evaporation of surface coatings can produce degradation in heat transfer surface properties and alter frictional effects. These problems will receive further discussion in the following section.

Vacuum

The pressure levels to be encountered by future lunar base systems will be expected to vary from sea level values on earth at launch to less than 10^{-13} atmospheres in space. Several deleterious effects which must be carefully defined can be expected from this high-vacuum environment. Seal leakage and cold welding are the most important of these.

Most lubricants are useless in space due to their high vapor pressure. Once lubricants and surface films are lost either by volatilization or as a result of frictional wear, the resulting uncontaminated surfaces may seize and cold weld. A leak-free method of transmitting motion into ultra-high vacuum thus is a major problem area.

The rate at which molecules leave a surface into a vacuum is given by the Langmuir equation:

$$W = (p/17.14) \ (M/T)^{1/2}$$

where W = rate of evaporation or sublimation, gm/cm^2 - sec.

P = vapor pressure of the material, mm. Hg.

M = molecular weight of the material in the gas phase

T = temperature, $^{\circ}K$.

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This equation gives the rate of evaporation or sublimation when none of the molecules leaving the surface returns to it. Calculated results for metals and organics and inorganics are given in the following section.

For short design periods, this effect is not considered serious, but for mission duty time of several years, sublimation and evaporation may degrade surface characteristics, particularly thermal control surfaces.

Lunar Surface Features

Due to the many uncertainties concerning the nature of the lunar surface, consideration of their effects is at best an intelligent guess. Most of the literature reporting on these effects is concerned with surface conditioning for passive thermal control and the possible nature of dust adhesion to thermal control surfaces.

For passive control for fuel storage, distribution of titanium dioxide (TiO_2) powder on the surface near the tankage has been proposed. Such a procedure could reduce the lunar surface radiation load and increase the effectiveness of heat dissipation of the tankage if proper values of (α/ϵ) can be obtained.

Results obtained from continued studies and the Ranger and Surveyor programs should provide clarification of the dust adhesion problem.

Environmental Effects on Materials

The following discussion outlines some of the effects of the space environment on metals, organics, and inorganics.

Metals

Mechanical Properties - The mechanical properties of metals should be affected by the lunar environment. Property data obtained in air may be used for design. The cryogenic properties of metals will have to be considered for applications where the sun is the only heat source, since temperatures in such cases will approach absolute zero during the lunar night. In general, tensile properties will increase with decreasing temperature, however, some metals exhibit brittle behavior, i.e., poor notch

sensitivity and ductility, at these low temperatures. Properties such as
thermal conductivity, expansion and electrical resistivity generally decrease
with decreasing temperature in the cryogenic range. Table 2-4 lists some
mechanical properties at low temperatures.

Surface Properties - Most metal surfaces will be quite stable in the high-
vacuum environment of the moon at normal operating temperatures. Cad-
mium, zinc, and magnesium are exceptions. Sublimation of these metals
at relatively low temperatures may be of concern if used as thin coatings,
or if there is a possibility of the metal plating out on cold surfaces acting
as electrical insulators. Only slight roughening of optically polished sur-
faces may occur with other metals. Table 2-5 lists sublimation rates of
some metals.

The frictional behavior of contacting metal surfaces poses a definite prob-
lem area. The small amount of data available indicates that similar
metals will seize when in sliding contact in high vacuum even under low
loads. There is a smaller probability that dissimilar metals will seize.
Anti-seize compounds and some dry film lubricants offer hope.

Electrical Properties - Metallic conductors should not be affected by the
lunar environment. Atomic displacements caused by the low-energy proton
particles from solar flares and steady solar emission, extend to a depth
of only 1 to 10 microns.

Organic Polymers

Mechanical Properties - The combined effects of high vacuum and high
energy electron bombardment during solar flare activity will definitely
affect mechanical properties of most organic polymers to some degree.
The process of degradation caused by the two damage sources differs.
High vacuum does not cause evaporation or sublimation of the surface of
polymers but initiates decomposition throughout the volume of the piece.
Electron particle radiation on the other hand creates damage to a partic-
ular depth, this depth being a function primarily of the material density.

TABLE 2-4

SOME MATERIALS PROPERTIES AT LOW TEMPERATURES

Material	Room Temperature					-350F				
	UTS (KSI)	YS (KSI)	Impact (Ft-Lbs)	Elongation (%)	MOD (PSI x 10^6)	UTS (KSI)	YS (KSI)	Impact (Ft-Lbs)	Elongation (%)	MOD (PSI x 10^6)
Aluminum Alloys										
1100-0	13	5	20			25	6	30		
2024-T4	66	46	5.5			88	62	6.5		
6061-TC	45	40	16.0			65	45	17.0		
Copper Alloys										
Berylco 25-1/2 H	100	96	72.0			115	110	60.0		
70/30 Brass	95	90	15.0			120	115	16.0		
Copper-annealed	32	9	52.0			55	12	65.0		
Stainless Alloys-Aust.										
304 annealed	100	35	56.0			240	50	60.0		
310 annealed	85	32	70.0			160	75	50.0		
321 annealed	100	60	50.0			230	60	30.0		
Nickel Base Alloys										
Inconel-80% cold Dr.	135	130	80.0			170	155	82.0		
Inconel X-Aged & Temp.	190	135	40.0			250	150	35.0		
Nickel	62	25	90.0			93	35	85.0		
Titanium Alloys**										
A-110-at-anneal	115	100	15			190	190	5		
B-120-VCA-Soln. Tr.	135	135	20			285	285	4		
Iron Base Alloys-Martensitic**										
410-1150F Temp.	180	140	55			235	210	5		
1075-OQ & Temp. at 720F	150	102	16			230	210	1		
Plastics*										
Kel-F-35% cryst.	5	5	1.4	140(.2 in.)	0.25	26	26	1.3	5(.2 in.)	1.15
Teflon-TFE-49% cryst.	4	2	4.0	450(1 in.)	0.10	18	18	0.8	2(1 in.)	0.60
Nylon-2.5% water	11	9	1.0	180(.2 in.)	0.20	35	35	0.3	1(.2 in.)	1.20

* No data found on other plastics. Embrittlement apparently precludes their use at cryogenic temperature.
** These materials are seriously embrittled by cryogenic temperatures.

TABLE 2-5

SUBLIMATION OF METALS AT HIGH VACUUM

Metal	Temperature at Which Given Sublimation Rate Occurs		
	10^{-5} cm./yr.	10^{-3} cm./yr.	10^{-1} cm./yr.
Cadmium	100 °F	170 °F	250 °F
Zinc	160	260	350
Magnesium	230	340	470
Lead	510	630	800
Manganese	840	1010	1200
Silver	890	1090	1300
Aluminum	1020	1260	1490
Beryllium	1140	1300	1540
Copper	1160	1400	1650
Gold	1220	1480	1750
Chromium	1380	1600	1840
Iron	1420	1650	1920
Nickel	1480	1720	2000
Cobalt	1500	1760	2020
Titanium	1690	1960	2280
Zirconium	2340	2740	3150
Molybdenum	2520	2960	3450
Carbon	2780	3050	4300
Tungsten	3400	3900	4500

Electron particle radiation damage will occur only during solar flare ac-
tivity and the depth of this damage can range from 0.0004" for low density
polymers like polyethylene to 0.0001" for higher density polymers like the
epoxies.

Table 2-6 lists data on the decomposition of some polymers in high vac-
uum. Considerable scatter in the temperature limit for 10 percent wt.
loss/year will be noticed. The reasons for this can be listed as follows:

1. Designations refer to polymers made from whole classes of monomers;
   each monomer or combination of monomers used can produce polymer
   with different high-vacuum resistance.

2. Small amounts of impurity can accelerate vacuum decomposition.

3. The incorporation of different additives such as catalysts, plasticizers,
   and mold lubricants can increase decomposition.

In general, weight losses of 1 or 2 percent do not produce property
changes of engineering importance, but losses of 10 percent create con-
siderable changes in engineering properties. Exceptions to this statement
must be made for any plastic or elastomer containing a plasticizer. Less
than 10 percent weight loss in plasticizer could embrittle the material
severely.

The case for plastics applications in the lunar environment looks black,
but this is partly due to the lack of specific information on degree of
damage. Polymers such as styrene, epoxies, reinforced resins (phenoloc,
epoxy, silicone, etc.) and elastomers such as kel-F, vinylidene fluoride-
hexa fluorpropane have shown some promise of retaining usable mechanical
properties.

Surface Properties - All that has been said concerning mechanical prop-
erties can be restated for surface properties with the problem of damage
from solar ultraviolet radiation added.

Of the two mechanisms of failure, ionization and atomic displacement,
ionization may be more important. Sunlight of the wavelengths below 500

TABLE 2-6

DECOMPOSITION OF POLYMERS IN HIGH VACUUM

Polymers	Temperature for 10% wt. loss/yr.
Nylon	80 -410°F
Epoxy	100-460
Urethane	150-300
Neoprene	200
Methyl methacrylate	220-390
Butyl Rubber	250
Styrene	270-420
Phenolic	270-510
Nitrile Rubber	300-450
Propylene	370-470
Natural Rubber	380
Silicone Elastomer	400
Mylar	400
Polyethylene (low dens.)	460-540
Polyethylene (high dens.)	560
Kel-F	490
Vinylidene fluoride	510
Teflon	710
Methyl phenyl silicone resin	>710

to 100 Å provides approximately 10^8 erg/cm^2-yr of ionizing radiation. Although the penetration ranges (i.e., the grams per square centimeter at which the flux is reduced by a factor e-base of natural logarithm), in the wavelength band 100-1000 Å are not clearly known; they are estimated at 10^{-4} -10^{-7} gm/cm^2. Clearly then, the resulting yearly doses would range between 10^{12} -10^{15} erg/gm. Such doses will cause severe damage to the properties of thin exposed layers of all known polymers. Changes in the optical properties reflectivity, absorptivity, and transmission - will be of importance for organic coatings for passive thermal control surfaces. Also, increased surface conductivity may require consideration for some applications.

Lubricants - Organic and Inorganic

Organic oils and greases will be unaffected by the radiation present on the moon; however, their resistance to vacuum and their lubricating properties under vacuum are questionable. A few of the problems associated with the use of oils in the lunar environment are listed below.

1. Low vapor pressure oils in general are not the best boundary lubricants even in air.

2. Load carrying capacities of petroleum oils are reduced considerably in the space vacuum.

3. Low vapor pressure oils have a high-pour point, and therefore many of these cannot be used at low temperatures.

4. In high vacuum, oils and greases may creep over clean surfaces and reach areas where they are not desired.

5. Fatty acid additives used normally to reduce friction depend on the presence of oxygen and water. Such additives would therefore not be beneficial in high vacuum.

Some of the diester and silicone oils and greases have provided lubrication in vacuum of 10^{-6} mm Hg for up to 1000 hrs at light loads. However, information for longer times and low temperatures is lacking.

More emphasis in testing has been put on dry lubricants and self-lubricating composite materials. Westinghouse studies have shown composite materials of Teflon reinforced with glass fibers and containing MoS_2 or WSe_2 powder and dry lubricants of MoS_2 and WSe_2, to exhibit low outgassing at temperatures up to 760F. Bearings of the 204 size 20-mm bore incorporating self-lubricating retainers made of Teflon-glass fibers - MoS_2, Teflon-bronze-MoS_2, and Teflon-silver-WSe_2, operated satisfactorily for 100 hours in a vacuum. $(10^{-7}-10^{-8}$ mm Hg).

Limited data on the aforementioned dry lubricants and self-lubricating composites indicate they function poorly at low temperature (-90F and below). The portions of rotating equipment requiring lubrication will probably require some heating if they are to be operated during the lunar night outside atmospherically controlled areas.

Inorganics

Mechanical Properties - If the temperature is not so high that appreciable sublimation occurs, there seems to be no serious evidence of any important decrease in mechanical properties of inorganic materials in vacuum, as compared to their properties under the atmospheres in which they are ordinarily used.

Surface Properties - Inorganic coatings are useful as thermal control surfaces and are less sensitive to radiation, high vacuum, and temperatures than organic coatings.

For relatively low temperature radiator surface applications, inorganics containing no volatile elements should not be seriously affected by decreases in optical reflectivity and increases in optical emissivity and absorptivity resulting from grain boundary sublimation and from differences in sublimation rates. However, sunlight in space affects primary optical properties. In particular, ultraviolet radiation in the region (2000-4000 Å), has experimentally been shown to degrade (increase) the solar absorptivity of several inorganic coatings. Also, many light-colored pigments used for thermal

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control surfaces are subject to color center formations which degrade the surface optical properties. This coloration generally decreases with temperature and also tends to bleach out on exposure to light which it can absorb.

Some controversy exists as to the nature of radiation degradation of surface coating for extended exposures. Telstar experiments indicate a definite degradation for only a portion of the exposure time, while other experimental studies indicate continued increases in solar absorption over the lifetime of the surface coating. Future experimental studies and space problems should help resolve this question and provide more quantitative data on the degradation of thermal control surfaces in the lunar environment.

C. RELIABILITY GUIDELINES

Reliability is a generic term frequently applied as a measure of the probability of a system, subsystem or component achieving the designated mission. Thus, reliability is inherently an established design objective which is verified through statistical environmental and performance testing within the range of application required by the mission. Numerous design techniques are employed in achieving desired reliability such as use of high-grade materials and parts, providing excess capacity, paralleling of components or systems, controlling environment, minimizing parts. In all instances, the system designer develops a trade-off scheme between gross weight and minimum time to catastrophic failure which can be accepted for the successful performance of the mission.

GENERAL GUIDELINES

The following general guidelines are applicable to all equipment which will be employed on the lunar surface:

An intensive effort is required to build reliability into all equipment to be used on the lunar mission; this may require the use of premium grade materials and parts. With the cost of transportation at $5000 per pound, heavy expenditures for reliability can be justified from an economic standpoint alone.

The degree of attained reliability must be demonstrated by reliability proof testing. Normal reliability prediction and assessment activities are to be conducted, but these are not to be used as a substitute for demonstration.

Within the general overall upgrading of the reliability of equipment, critical components are to be selected for concentration of effort. In any system it is possible to give priority to only a limited number of items.

Redundancy will be used where necessary, but it is not to be used as a substitute for building inherent reliability into a part, assembly, component, or subsystem.

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Requirements for preventive maintenance are to be held to the minimum. This reduces the possibility of human errors and also conserves manpower.

Provision is to be made for repair by replacement of selected parts, assemblies or components. However, this will not be used as a substitute for inherent reliability of parts, assemblies or components, nor for the omission of necessary redundancy.

All parts which are to be replaced on schedule as a planned part of the reliability or life requirements are to be designed to prevent improper installation through human error.

MISSION REQUIREMENTS

Each element of a total mission must be assessed a reliability penalty. Those elements which cause an abortion of the mission are required to demonstrate very high probability of success, whereas those elements which cause a postponement of a less important sub-mission can be permitted to have lower levels of reliability. Thus the most important aspect in establishing "reliability guidelines" for the Multipurpose Engine Fuel System Concept is to create a model of the anticipated missions and sub-missions and to assign reasonable minimum times to catastrophic failure for each. Thus, when a given system cannot meet this minimum, an assessment of additional weight must be made to bring its probability of achievement to the required level. The purpose of this guideline is to establish minimum requirements for each mission and sub-mission of the Lunar Base where the Engine-Fuel system may be utilized and to cause all candidate systems to be measured on the same ground.

Lunar Base Missions are postulated to include the following: Support of man against the hazards of the lunar environment. This occurs in the following principal areas:

1. Personnel shelter
2. Manned lunar vehicles
3. Manned central base activities

Support of base construction. This occurs in the following principal areas:

1. Unmanned construction equipment
2. Communications

Support of scientific exploration. This occurs in the following principal areas:

1. Astronomical observatory
2. Selenological laboratory
3. Materials and process laboratory
4. Command and communications center for future unmanned exploratory vehicles.

A mission abort will be defined as the returning to earth of the astronauts prior to the scheduled time. Thus any return necessitated before the completion of a 90-day mission for Phase I of the Lunar Base or before a scheduled close down and return to earth for subsequent phases of Lunar Base will be considered a mission abort. Only one application can be considered to offer such potential - the power system associated with providing the environmental control and life support in the shelter modules. Thus, for example, the minimum continuous operating time which can be accepted for the shelter power system for Phase 1 is 2160 hours, which is rounded off to 2500 hours to provide a reasonable safety factor.

The next most critical mission occurs when a man is operating separated from the central base in a lunar vehicle. Here the power system must have the capability of permitting man to survive a mobility failure and return to the central base by means of a "rescue" vehicle dispatched at his command. Thus the vehicle power system must be capable of providing environmental

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control and life support for a time period equal to 150 percent of the trip
time when the trip time is a variable in each base phase.

The next most critical mission occurs when a base construction or scien-
tific mission must be deferred due to power system failure. This failure
is less critical than loss of life support power, and one can therefore ac-
cept lower reliability for such power systems. A rough estimate shows
that only about 25 percent of the total available time will be available for
work requiring auxiliary power. Thus the portable power systems will be
assessed a reliability requirement of 500 hours continuous operating time.

SCHEDULED REPLACEMENT

The method of scheduled replacement similar to that employed on aircraft
engines will be used to insure that power is available for critical missions.
As indicated above, these missions are life support and environmental con-
trol in shelter modules and operation of manned lunar vehicles. In power
systems planning for these critical missions, a reasonable equivalent full
power operating life for the engine and fuel system will be determined, which
can be demonstrated by reliability proof testing with a high degree of con-
fidence. A sufficient number of power units will be scheduled for the par-
ticular mission such that their total demonstrated operating life will equal
the mission time requirements. In the case of power for the shelter mod-
ules, two or more units will be connected to the system. Unit No. 1 will be
placed on line for its demonstrated life at which time it will be placed in
standby, and Unit No. 2 placed on line and so on until the end of the mis-
sion. In the case of manned lunar vehicles, Unit No. 1 wiil be removed and
replaced by Unit No. 2 at the end of demonstrated life, or sooner in case
the vehicle is going away from the base and a scheduled change would occur
before return. Scheduled replacement also will be used for portable power
sources used away from the base. However, for portable power systems
and unmanned vehicles used around the base, the engine and fuel systems

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will be operated to failure before making replacement. An example is given below on the power system for environmental control and life support for the shelter module.

POWER SYSTEM FOR SHELTER MODULE

The reasonable expected minimum time to failure for any of the engine and fuel systems under consideration is 1000 hours, which is less than half the required continuous operating time for the power system associated with providing environmental control and life support for the shelter module for Phase 1. In order to provide the required 2500 hours of continuous operation, three units will be scheduled so that they can be operated consecutively, i.e., Unit No. 1 will be operated 1000 hours; then it will be shut down and Unit No. 2 placed on line for the next 1000 hours; and finally the third unit will be placed in service. Units removed from line at the end of 1000 hours will be in stand-by status. If a reliability requirement of 1000 hours minimum time to failure with a probability of 95 percent is established, a mean time between failure (MTBF) of approximately 20,000 hours must be demonstrated. The demonstration test will require the operation of 20 units for about one year, which is considered to be a reasonable figure.

SELECTION CRITERIA

The requirement that power systems have a specified minimum time to failure with a specified probability, which can be demonstrated by reliability testing within reasonable economic limits, will be one of the criteria for choosing the concept for the engine and fuel systems. Each of the systems under consideration will be weighed against the criteria established for the particular use which is involved.

D. COMPARISON TECHNIQUE

The effort this month was directed primarily at a "first pass" discrimination of hypothesized systems resembling a number of actual engine and fuel systems that might be considered for lunar base application. The systems considered are indicative of static, dynamic, propellant-consuming, and solar source systems. Also, this approach illustrated a technique for subsequent system analysis. The systems hypothesized and the engine and fuel system that resembles these schemes are:

System	Resembles
A	- Fuel Cells
B	- Solar Turbine and Batteries
C	- Solar Turbine for Daytime Use and Fuel Cell for Night
D	- Solar Cells plus Batteries
E	- Internal Combustion Engine
F	- Solar Thermoelectric Generator plus Batteries

When specific definitions for missions are available, this method of analysis will serve as a basis for comparison. In the absence of defined mission requirements, these systems were compared on the basis of total cost (fixed plus operating) per lunar day, for the specific load profiles presented in the September Monthly Report (WANL-PR(S)001-B); namely:

Load	Peak Power	Load Factor (average power/ peak power)	Use Factor (% of lunar day that plant is operating)
Shelter	5 Kw	0.75	0.9
Mobile Vehicle	5 Kw	0.75	0.2
Mobile Vehicle	35 Kw	0.75	0.2
Auxiliary Supply	5 Kw	0.9	0.1

The results of this hypothesized system comparative analysis are presented in Figure 2-18. The following table gives a listing of system and subsystem parameters used in obtaining the results displayed in Figure 2-18.

TABLE 2-7

System A   (like Fuel Cells)

Weight of Complete System	= 400 lb/kw
SPC	= 1#/kwh
EFPH*	= 1000 hrs
Propellant weight/Total weight	= 0.8

System B   (like Solar Turbine & Batteries)

Turbine Weight	= 250 lb/kw
Turbine EFPH	= 1000 hrs
Turbine Efficiency	= 20%
Battery Weight = 40 lbs/kwh	= 25 watt-hrs/lb
Maximum Discharge	= 50%
Battery Life	= 1000 cycles (12 cycles/yr)
Solar Concentrator Weight	= 10 lbs/ft^2
Concentrator Efficiency	= 90%
Target Efficiency	= 60%

Coupling these efficiencies with
turbine efficiency means 70 ft^2/kw, or 700 lbs/kw

Concentrator Life	= 3 years

System C   (like Solar Turbine, Fuel Cell Combinations)

Fuel Cell as defined in System A for night-time operation.

Solar Collector, plus turbine, as specified in System B, are used during daytime.

Battery capacity taken at 10% of that required in System B.

System D   (like Solar Cells plus Batteries)

Solar Cell Weight	= 200 lbs/kw
Solar Cell Cost	= $10/cell
Number of Cells/kw	= 80,000
Battery - Same as for System B	

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*EFPH = equivalent full power hours.

TABLE 2-7 (continued)

System E (like Internal Combustion Engine)

Internal Combustion Engine Weight	= 150 lbs/kw
SPC	= 1.25 lbs/kwh
Engine EFPH	= 1000 hrs

System F (like Solar Thermoelectric Generator, plus Batteries)

Thermoelectric Generator wgt. (complete)	= 800 lbs/kw
Thermoelectric Generator EFPH	= 1000 hours
Generator Thermal Conversion Efficiency	= 5%
Solar Concentrator Weight	= 2800 lbs/kw
Other Concentrator Parameters	= same as for System B
Batteries	= same as for System B

The individual bars in Figure 2-18 are subdivided into the component costs based on a launch cost of $5000/lb. The cost component ascribed to capital expenditure, such as turbine, fuel cells, battery, etc., is obtained by multiplying the lunar delivered cost by the fraction of its useful life that is used per lunar day.

Although the results shown in Figure 2-18 are by no means final, they lead to a number of interesting considerations, some of which are expected to be useful during subsequent phases of this study.

There is a marked difference in the total cost per lunar day for the different systems hypothesized in Figure 2-18. Therefore, it is possible to select specific engine and fuel systems for further study while eliminating others without the need to obtain the hyperfine detail. At this stage, the accuracy of input data permits selection if factors of two or more exist; if less separation exists, more accurate data is required. For example, System D costs per lunar day are almost a factor of 4 less than System E.

Assuming that System A, which resembles the H_2-O_2 Fuel Cell, were to have a 100% conversion efficiency, then for the shelter profile the propellant cost would be $9.6 million per lunar day instead of the $15 million. However,

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even at 100% fuel conversion efficiency, System A is about three times more costly per lunar day than System D. This consideration is significant, since presently fuel cells are often mentioned as the preferred system.

With the exception of System A, the assumed parameters used in obtaining the results shown in Figure 2-18 are conservative when compared to present literature, government agency, and industry identified values. The following illustrates this:

Component	Quoted Values	Value Used in Table 2-7
Batteries	25-40 watt-hrs/lb.	25 watt-hrs/lb.
Discharge	70-80%	50%
Solar Concentrator (System B)	300 lbs/kw	700 lbs/kw
Thermoelectric Generator	400 lbs/kw	800 lbs/kw
Turbine Weight	150-200 lbs/kw	250 lbs/kw
Fuel Cell	400 lbs/kw	400 lbs/kw

The analysis of Figure 2-18 has used the most favorable values for the fuel cell capability while being somewhat more conservative with the other systems.

The preferred system for all of the cited load profiles, if cost be the criteria for selection, would use solar energy rather than earth-produced, missile-transported propellant.

In all of the systems considered in Figure 2-18, with the exception of System D (solar cells), the cost of the hardware is negligible compared to the launch cost. However, with solar cells at $10/cell, the hardware cost is about $5000/lb, which is comparable or equal to the assumed launch cost.

The September Monthly Report showed that a typical H_2-O_2 regenerative fuel cell would require about 5 kw (elec) input from a nuclear plant to produce 1 kw (elec) from the fuel cell. Assuming that the nuclear power plant will

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1. Weigh 250 lbs/kw (electric)

2. Have a three-year useful life at an 80% load factor,

then System A for shelter application is reduced to about $5.3 million per lunar day rather than the $19 million when propellant is required. The above example neglects the cost associated with reactor development and flight hardware. This illustrates that a regenerative system such as System A can be in the same "ball park" with solar energy using Systems B and D.

Table 2-7 indicates that most of the engine types thus far considered have an EFPH of 1000 hours and therefore, depending upon lunar day load characteristics, may require relatively frequent replacement. For example:

Load	Replacement Time (Lunar Days)
Shelter	2.1
Mobile Vehicle	9.3
Auxiliary	15.4

During the coming month, additional engine and fuel systems such as SNAP devices and lithium hydride thermal storage, coupled with a solar energy system are scheduled to be analyzed in terms of cost per lunar day. Also, as the techno-economic parameter estimates discussed in the previous monthly report become available, it should be possible to do overall system comparison based on a figure of merit that includes, with appropriate sensitivity, all the factors for final system selection. This method of analysis will be extended to determine the sensitivity of the important characteristics of the systems and combinations of system in the figure of merit expression. Modifications will be made in the basic equation, so that appropriate weighting of the factors will more closely relate to the value of each system for specific missions.
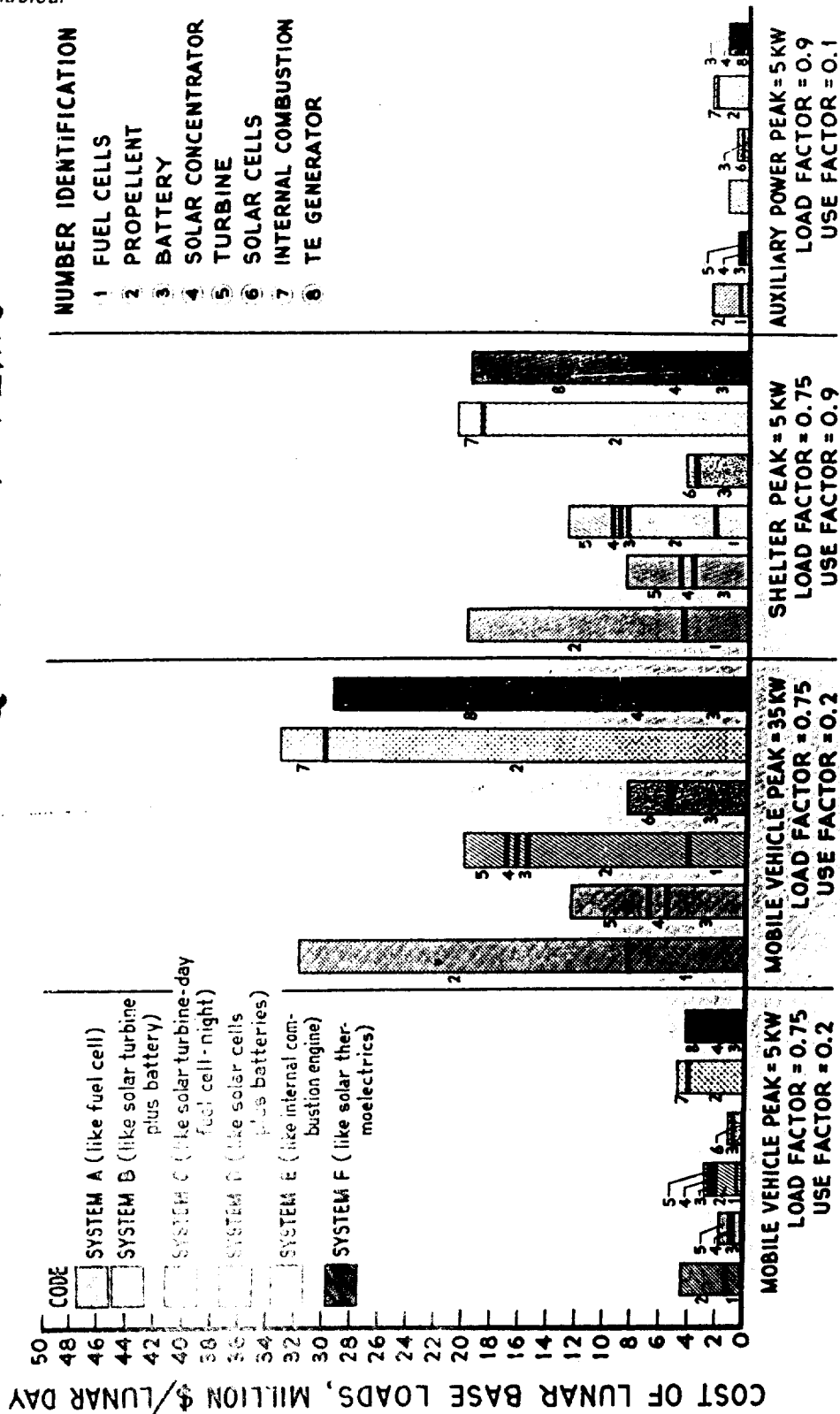
Figure 2-18. Load Profiles

E. COMPARISON CRITERIA

GENERAL REQUIREMENTS

Applications

The Multipurpose Engine and Fuel System (MEFS) must be capable of furnishing power for all energy consuming loads in the lunar base, except for those that are directly supplied by the lunar nuclear prime power plant. The applications will include:

1. Vehicular propulsion, 1 - 100 kw with emphasis on the 1 - 30 range
2. Shelter power source, 0.5 - 4 kw
3. Portable power source, 0.5 - 20 kw

System Evolution

The lunar engine and fuel system must be capable of evolution through the following phases:

1. Engines for incorporation into developmental lunar vehicles and equipment must be available two years before base initiation. A lunar refueling requirement is not necessary at this time.
2. A simple lunar refueling system which is completely dependent on re-supply of fuel from earth must be available on base initiation.
3. A second generation fuel system which provides some degree of logistic independence must be available two years after base initiation. Nuclear prime power will be available.
4. A fuel system which provides the highest degree of logistic independence must be available three years after base initiation.

System Loads

For the purpose of comparing system concepts, the following power demand guide will be used:

	Mobile Vehicle Power Supply 1 - 100 kw		Shelter Power Supply 0.5 - 4 kw	Portable & Misc. Power Supply
Capacity (kw)	5	35	5	5
Operation Factor	0.2		0.9	0.1
Load Factor	0.75		0.75	0.9
Power Type	4 Motive 1 Electrical	30 Motive 5 Electrical	Electrical	Electrical
Unrefueled Time (hrs)	24 - 100	1000	24 - 100	24 - 100
Use	Mobile		Stationary	Portable

The total output of the engine and fuel system is assumed to be 10 kilowatts initially, and will increase to 200 kilowatts within two years.

Utilization of Materials

Life Support and Propulsion Materials

The fuel type shall be selected considering the over-all benefits to the lunar base. Fuels which can also be used in life support or as a spacecraft propellant will receive primary consideration. Materials which can be used in life support and as fuels are oxygen, hydrogen, water and nitrogen.

Lunar Materials

Fuel material which can be obtained by extraction from the lunar crust will also be given special consideration. These materials are assumed to be limited to water, which may be found as hydrates, and chemically bound oxygen.

Salvage Materials*

There is a possibility that fuel, useful tankage and other salvage equipment from expended spacecraft will be available. The salvage fuel type considered will be limited to the Lunar Excursion Module fuel, Hydrazine,

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*Indicates revision or addition of detail.

UDMH, Nitrogen tetroxide, and to hydrogen and oxygen, either liquid or gaseous. The total quantity of available fuel from each spacecraft will be assumed to 1500 pounds. The fuel/oxidizer ratio for H_2 - O_2 will be assumed as 1:5.

Packaging

Payload Restraints

All equipment items and fuel must conform to the "Payload Restraints" as described in Annex A. Detailed criteria will be established later.

Unattended Storage

Components of the engine-fuel system must be capable of remaining on the lunar surface for an unattended storage period of up to one year. Equipment performance must not be seriously degraded during this period. Detailed criteria will be established later.

Modular System

All system components shall be designed on a modular basis. Modules will perform a distinct, integrable function. The system will be capable of growth by the addition of modules. Individual modules will have the payload weight and volume restrictions as noted in "Payloads Restraints", Annex A.

Logistic Advantage

Regeneration and Power Requirements

The regeneration concept will yield a net logistic advantage considering the weight and component life of the regeneration equipment. That part of the nuclear power plant weight charged to the fuel regeneration system will be included.

Lunar Resource Extraction Requirements

The lunar resource extraction concept will yield a net logistic advantage considering the weight, component life and fueling requirements of the extraction equipment. That part of the nuclear power plant weight charged to the extraction process system will be included.

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Base Compatibility

The engine and fuel system concept must be compatible with complementary and concurrent work on the lunar base being conducted by other facilities. Definite requirements in this will be noted as they arise.

Development Time and Costs*

The selected system should be capable of test flights in 1968, and be operational in 1970. Materials feasibility must be shown by 1963.

The development costs should be reasonable, and maximum consideration should be given to those systems which can use components or subsystems that are already under development.

Environmental Requirements*

The environmental restraints detailed in Annex B, C, and D of the letter request will be used. These annexes detail the meteoriod, radiation, and lunar surface environments. The environmental assumptions will be as follows:

1. Temperature — $120^{\circ}K$ to $380^{\circ}K$
   (Lunar Surface)

2. Pressure — 1×10^{-13} earth atms.

3. Gravity — 1/6 earth gravity

4. Meteoroids

   a. Armor protection only will be used.

   b. A probability of no punctures in three years at a level of 0.99 will be used.

   c. The shield penetration phenomenon is defined by the Ames (Charters and Locke) criterion with an uncertainty factor of 5 included to allow for any variations from this criterion.

   d. The meteoroid spectrum is limited to spherical particles having a density of 0.5 gm/cc and a velocity of 30 km/sec.

   e. A meteoroid will penetrate a thin wall having a thickness 1.5 times the radius of the hemispherical hypervelocity crater.

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*Indicates revision or addition of detail.

f. The meteoroid flux is given by the equation

$$N = 4.4 \times 10^{-14} + 0.4 \, M$$

in which M is the visual astronomical magnitude of a point source and N is the cumulative flux of meteors brighter than magnitude M.

g. The initial kinetic energy of a meteor is defined by the equation (in MKS units)

$$mv^2 = 3.7 \, Lv^{0.3} \times 10^{4 - 0.4 \, M}$$

in which L is a dimensionless constant assigned a value of 2.78 to make the zero magnitude 30 Km/sec. meteoroid weight 2.5 grams.

5. Radiation - to be detailed later

Performance Requirements

Weight and Volume

Engine and fuel regeneration system components must have a high extractable energy for a given system weight and volume.

Reliability*

All components should have the highest practical reliability. Considering the limited performance history of many of these components, good engineering judgment will be used where necessary.

The following specific reliability requirements will be used in concept comparison:

1. Reliability must be demonstrated by test.
2. Preventive maintenance procedures are to be held to a minimum.
3. Scheduled replacement will be used to achieve reliability.
4. Redundancy will be used to achieve reliability levels when required.

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*Indicates revision or addition of detail.

Astronuclear

5. Shelter engines must be capable of 2500 full power operating hours.

6. Mobile engines must be capable of providing life support for a time period equivalent to 150 percent of the trip time (vehicle range).

7. Portable engines must be capable of 500 full power operating hours.

8. Regeneration plant requirements will be detailed later.

Adaptability

The selected system will have a wide degree of applicability. Preferably a single engine type will be selected that will accommodate all expected loads, regardless of location, size, and power type requirements whether electrical, motive, or thermal. Similarly, a single fuel type is preferred so that logistics and regeneration problems will be lessened.

Storage

Fuel storage requirements will be detailed later.

Operational Requirements

Refueling

Engine systems must be capable of being refueled or recharged whether the application type is mobile, stationary or portable.

Day-Night Operation

Energy conversion systems must be capable of supplying all demand loads during both the lunar day and lunar night. Loads will be both continuous and intermittent. Equipment must be capable of restart after extended shutdown periods.

Personnel

Reasonable demands can be made on lunar base personnel for maintenance, operation, and installation needs of the engine-fuel system. However, the mission requirements of the lunar base personnel take precedence over these support functions. The system will not require highly specialized personnel for these functions.

Control

The control mode may range from completely earth-controlled to control at the operation site, and from completely automatic to manned operation. Combinations within these ranges will also be considered.

III. ENGINES

A. SUPPLIER VISITS

COMPANY: ALLIS CHALMERS* (Milwaukee), September 20, 1963

Names of Participants

For Allis Chalmers: Dr. Powel A. Joyner, Assistant Director, Research Division; Dr. Patrick G. Grimes, Senior Research Chemist, Research Division; R. L. Kamber, Marketing, Research Division; David Ghere, Development Engineer, Research Engineer.

Type of Engine: Fuel cell

Fuel: Hydrogen and Oxygen (Also Hydrayine, Hydrocarbons)

Engine General Description

A medium temperature (= 200°F) aqueous potassium hydroxide solution is contained by capillary action of an asbestos membrane. On either side of the membrane are porous nickel electrodes which are supported by nickel plated magnesium holders which also manifold the reactant gases and form the conductive electric path. On the H_2 side of the membrane the product water vapor is formed and is removed through a KOH saturated membrane under the action of an applied pressure differential.

COMPANY: GENERAL ELECTRIC* (Lynn), October 9, 1963

Names of Participants

For General Electric: Marvin P. Eisen, Senior Systems Engineer; Richard Blackmer, Advanced Engrg., Gemini Program; Paul Schratter, Gemini Program.

Type of Engine: Fuel cell

Fuel: Hydrogen and Oxygen

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*Additional information, proprietary in nature, is in Section V.

Engine General Description

An ion exchange membrane fuel cell operating at approximately $130^\circ F$. The membrane is flanked on either side by nickel electrodes, which in turn are separated by a manifold member which distributes O_2 and H_2 and contains cooling tubes and a fibrous wick for moisture removal. The product water vapor is condensed directly in the cell and is removed by capillary action. The current passes perpendicular to the plane of the membrane, electrode, and manifold thereby directly connecting the units in series.

COMPANY: PRATT AND WHITNEY, E. Hartford, Conn., October 10, 1963

Names of Participants

For Pratt and Whitney: J. W. Connors, Systems Analysis; Richard Vasques, Marketing; Dr. Richard Briceland.

Type of Engine: Fuel cell

Fuel: Hydrogen and Oxygen

Engine General Description

This fuel cell is commonly known as the Bacon Cell and uses molten potassium hydroxide (85 percent solution) operating at about $450^\circ F$ as the electrolyte. The liquid electrolyte is contained on either side by dual porosity nickel electrodes. The purpose of the dual porosity is to maintain the liquid gas interface at a given location (near the plane of the porosity change), by means of a small pressure difference which is resisted by capillary forces. The water produce is formed as a vapor on the hydrogen side of the cell, and the mixture is recirculated through a heat exchanger which condenses the water and cools the hydrogen, which in turn cools the fuel cell.

Development Program

1. Status: Apollo program. Data not yet available.
2. Level of effort: Apollo fuel cell program in excess of $30,000,000.
3. Problem areas: Information not yet available.

Performance

1. Applicable power range: Very wide. Depends upon mission.
2. Output power type and control: Dc power normally at 28-30 v, but can be built for higher or lower voltages. Voltage control is a function of load variation and varies from 0.85 v at 450 amps/ft^2 to 1.2 v at 0 amps/ft^2. In normal operation the power range, and therefore the voltage range, is much less than this. Figure 3-1 shows percent voltage change versus percent power.
3. Weight: Information not yet available.
4. Volume: Information not yet available.
5. Propellant consumption rate: Information not yet available in complete detail. However, Figure 3-2 shows the propellant consumption of a typical fuel cell which is not defined as regards weight and operating conditions.
6. Heat rejection:

   <u>Quantity</u> - Information not yet available in detail. Attached Fig. 3-3 shows flow of hydrogen coolant.

   <u>Temperature of engine coolant</u> - Hydrogen and water vapor are circulated from the fuel cell to the radiator (or other heat exchanger). The hydrogen and water vapor is about 450°F coming from the fuel cell. The temperature of the mixture must be reduced sufficiently to condense the water vapor and cool the hydrogen enough so that it can cool the fuel cell. Fig. 3-3 shows the relationship between hydrogen relative flow rate and its temperature after return to the fuel cell as a function of percent maximum power. Absolute values are not yet available.
7. Cooling capacity: No net cooling capacity.
8. Starting:

   <u>Hardware and Procedure</u> - Information not yet available although it is clear that the fuel cell must be heated initially to some temperature so that the reaction will start.
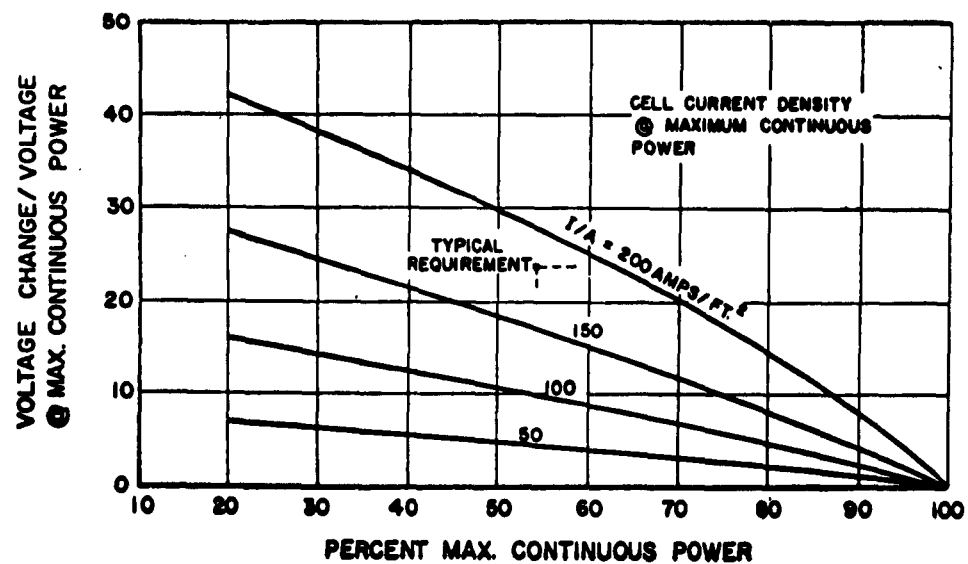
Figure 3-1. Cell Voltage Regulation
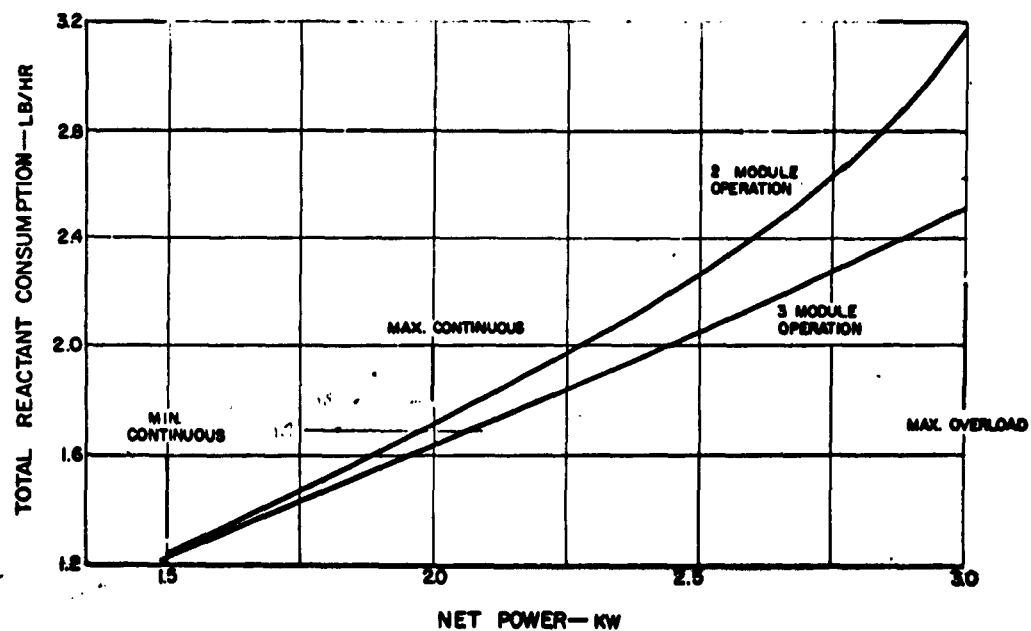
Figure 3-2. Oxygen and Hydrogen Consumption

CELL TEMPERATURE 500°F
CONSTANT HYDROGEN INLET SPECIFIC HUMIDITY
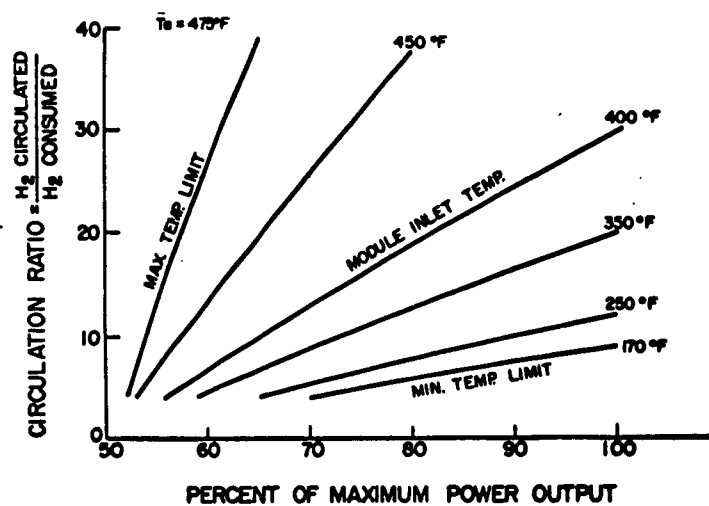
Figure 3-3. Hydrogen Circulation Required for Waste Heat Removal

Power level - Power is a function of starting time, insulation, and environment.

9. Overload Characteristics: See Figure 3-4.
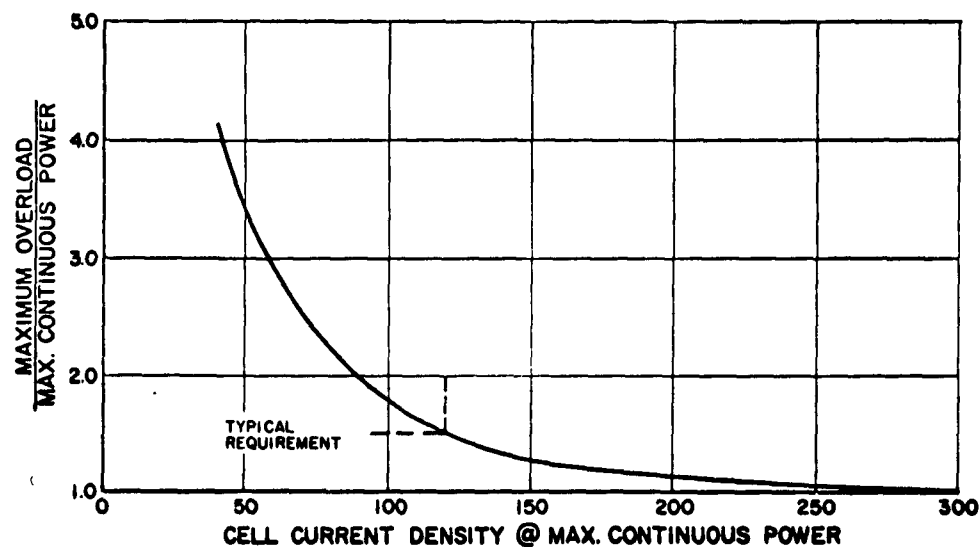
Figure 3-4. Overload Capabilities

10. Exhaust products: Liquid water.

11. Ruggedness: Information not yet available, although Apollo application presumably must satisfy launch specifications.

12. Effect of environment: Information not yet available.

13. Life: Information not yet available.

14. Storability: Information not yet available.

COMPANY: WALTER KIDDE, Belleville, New Jersey, October 11, 1963

Names of Participants

For Walter Kidde Co.: K. Traynelis, Associate Tech. Director; G. Hall, Project Engr.; E. Gedder, Product Mgr.

Type of Engine: Integrated Cryogenic Dynamic Power System
Fuel: Hydrogen and Oxygen

Engine General Description

Cryogenic hydrogen and oxygen are expelled from storage tanks by means of heat addition. The hydrogen is passed through a heat exchanger, where it absorbs heat from relatively low temperature sources such as electrical and electronic equipment, etc. The hydrogen then passes to another exchanger where it exchanges heat with the oxygen prior to their admission into a catalytic burner or gas generator. The mixture of hydrogen and water vapor is then expanded through a multiple reentry staged axial flow impulse turbine. The turbine drives an alternator which provides the output power.

Development Program

1. Status: A breadboard system has been built and tested by Kidde and has been delivered to the Air Force for further development.
2. Level of effort: Kidde has completed its contract and is not doing further work at this time.
3. Problem areas: Life, fuel consumption improvement, system operation and control.

Performance

1. Applicable power range: Above about 10 kw.
2. Output power and control: The alternator runs at 12,000 rpm. Frequency and voltage level is a design choice. Voltage control can be made very precise if required.
3. Weight: 119.8 pounds for a 25 hp unit. That is, 4.8 lb/hp or 6.42 lb/kw.
4. Volume: Function of hp level. At 25 hp the volume is approximately 1 to 2 cubic feet in the flight version.
5. Propellant consumption rate: See Figure 3-5
6. Heat rejection.

    No net heat rejection
7. Cooling capacity: Using hydrogen only as a heat sink, and starting with subcritical liquid, the cooling capacity at 25 hp with current tested SPC is:
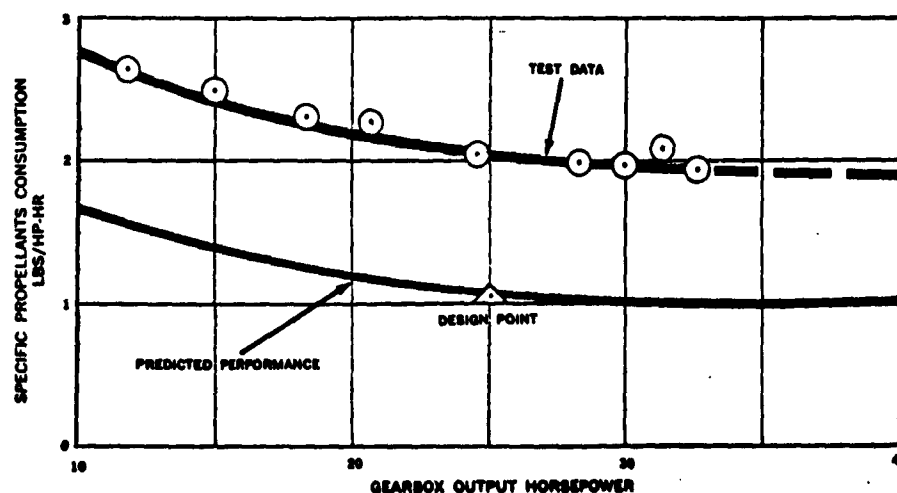
Figure 3-5. Four Stage Hybrid Turbine Performance Using Hydrogen -
Oxygen Propellants

$$2.1 \frac{lbO_2 + H_2}{HP \ HR} \quad x \quad 0.862 \frac{lbH_2}{lbH_2 + O_2} \qquad x \quad 2200 \frac{BTU}{lbH_2} \ = \ 3,990 \frac{BTU}{HP \ HR}$$

@1800°F

The cooling capacity at 25 hp with projected SPC is:

$$1.1 \frac{lbH_2 + O_2}{HP \ HR} \quad x \quad .862 \frac{lbH_2}{lbH_2 + O_2} \qquad x \quad 2200 \frac{BTU}{lbH_2} \ = \ 2.085 \frac{BTU}{HP \ HR}$$

@1800°F

8. Starting:

Hardware and Procedure - Not detailed for flight system. Probably a
battery source for heating the propellants and operating the heat exchanger
hydrogen recirculation blower, for breadboard test, gaseous hydrogen and
oxygen were used for startup.

Power Level - Not known. Relatively small quantity.

9. Overload characteristics: Breadboard system was tested at 32.5 hp, which
is 30 percent overload. Momentary overload could probably exceed this
substantially.

10. Exhaust products: Mixture of water vapor and hydrogen in the ratio of approximately $\frac{.18 lb\ H_2O}{lbH_2}$.

11. Ruggedness: System consists of turbo-machinery and heat exchangers which have inherent good shock load capability.

12. Effect of environment: Since system exhausts to ambient, on the moon, the exhaust port would have to be sealed against the vacuum when not in use.

13. Life: System has not been engineered for long life to date. At current design turbine inlet temperature of 1800°F, the turbine is probably the limiting factor.

14. Storability: No specific problems foreseen.

THE MARQUARDT CORPORATION, Van Nays, California, October 16, 1963

Names of Participants

For Marquardt: M. E. Goodhart, Mgr., Advanced Technology Development; Andy Duncan, Assistant to Manager.

Marquardt has been working on internal combustion engines using hydrazine and N_2O_4. The latest contract was for the purpose of documenting the performance of a bench type I. C. engine. The present effort is in closing out this contract.

Most of the trouble in this engine has occurred in the injection system, although piston rings have also failed. The power levels that have been considered are 2.5 kw and 6 kw average shaft power. The output power would normally be 110/200 volt ac, 400 cps from an alternator driven at 4000 rpm by direct coupling.

At 2.5 kw the specific weight will be $15\frac{lb}{kw}$ and at 6.0 kw it will be $10\frac{lb}{kw}$ not including the alternator and heat exchanger weight. The specific propellant consumption could be as low as 4.0 lb/kw hr at the shaft, but this has not yet been documented. About 25 percent of the total energy input must be rejected from the cylinder by a coolant running at 200°F, another

25 percent is discharged in the exhaust. Thus maximum shaft output efficiency is 50 percent. Preheating must get the lubricant warm enough for satisfactory lubrication of the engine. The engine must be motored to about 400 rpm for startup.

At present the engine speed is 4000 rpm with control to ±200 rpm. The power profile for the LEM was used for the design of the 2.5 kw engine.

Contamination of the lubricant with the propellant or exhaust products causes lubrication failure. This is not considered a serious problem for missions presently under consideration (<240 hrs). Derating to obtain 2000 hr MTBF is likely to increase SPC by a factor of 2 (8 lbs/kw hr shaft). Storability is only limited by the lubricant, coolant or propellant. Very little is known about the life but it is expected to be of the order of hundreds of hours. A materials compatibility problem exists in handling hydrazine and N_2O_4 in the fuel injector system. Presently used material is 440-SS with surface diffused chromium to a Rockwell C72 hardness.

The current status of development does not indicate promise for use in the early phases of the Lunar Base program except for special singed use applications.

G. M. DEFENSE RESEARCH LABORATORIES*, Santa Barbara, California, October 16, 1963

Names of Participants

For General Motors: Dr. Friedman, Head, Land Operations and Lunar Program Mgr.; S. Romano, Head, Lunar Program Office; C. R. Russell, Power Systems; P. D. Agarwal, Head, Electric Power and Propulsion; N. D. Miller, Head, Concept Analysis; J. Finelli, Mobility Laboratory; F. Pavlics, Mobility Laboratory

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*Additional information, proprietary in nature, is in Section V.

The visit to G.M. Research Laboratories was for the purpose of obtaining information to assist in determining the characteristics of a vehicle E-F system.

A presentation of the articulated multiple element vehicle was made. This concept evolved from the G.M. position of pushing conventional vehicle design to the limit of the state of the art.

They demonstrated a maximum mobility vehicle concept for an assumed lunar surface. Further exchange of information would be possible at a future date if thought necessary.

VICKERS, INC., Torrance, California, October 17, 1963

Names of Participants

For Vickers: Norman Morgan, Project Engineer, Aero Hydraulics Div.;
William A. Bass, Project Engineer, Advanced System Analysis.

Vickers, Inc. has been developing a hydrogen rich internal combustion engine and has proposed a stoichiometric internal combustion engine.

The fuel rich I. C. engine has been developed through the bench model hardware stage. The biggest development problem to date has been materials and life of the oxygen valve.

These engines appear to be most applicable in the 1 to 10 kw range, where alternators or hydraulic pumps are to be driven as energy converters. With the alternator, a 2 kw engine system less heat rejection hardware and fuel system would weigh 85 lb. Ultimate specific weight would approach 15 lb/kw. The volume of such an engine-generator assembly would be less than 0.5 ft^3.

With the propellants supplied at the useable pressure the present corrected propellant consumption is 1.8 lb/kw at the shaft. It is expected that this will be reduced to 1.35 lb/kw.

Heat rejection is to a cooling jacket at 200°F with the heat load equal to about 140 percent of the shaft power. The exhaust products are at 140°F and

are only partly recoverable. The amount of cooling capacity for the super-critical storage of propellants per kw-hr of output would be 1900 btu/kw hr up to room temperature.

Starting requires preheating and motoring to 800 rpm. Speed control can be to ±1/2 percent. SPC will vary less than 10 percent over the full range of operation.

The lower temperature limit for storage of the dry engine is set by the lubricant, which also fixes the required preheat. Long term storage does not appear to be a problem. Nitrogen is suggested as storage gas and may require 15 sec to purge before startup. Meteoroid, vacuum, dust and radiation considerations are generally overcome by the basic design requirements, and the effects of these are included in the weights mentioned.

Life of this type of engine is predicted by Vickers to be in the thousands of hours. The only demonstrated performance has been 50 hours of total run with only 1 hour of continuous operation. They claim continuous run is limited by their hydrogen supply.

A proposed stoichiometric H_2-O_2 I. C. engine operates in a similar manner, but includes water injection to provide a diluent to keep peak cycle temperatures below 5000°F.

This engine is still in the initial design stage, but no new problems are anticipated.

The objective specific propellant consumption for this design is 1.0 lb/kw at the shaft. An estimate of the engine specific weight is 18 lb/kw.

For a fully condensing cycle the heat rejection will amount to approximately 0.75 the power output. The temperature of this heat rejection will be greater than 130°F. The cryogenic cooling capacity to room temperature for this engine will be small - about 400 btu/kwe hr for subcritical storage.

Startup is not considered a problem, but preheating and motoring will be necessary. The characteristic flat propellant consumption curve would probably be applicable to this engine.

The mechanical integrity of this, as well as any reciprocator, would probably exceed the lunar environment design considerations of meteoroids, dust, radiation and temperature. A life comparable to the life of the fuel rich I. C. engine could be expected. A typical 25 hp unit would be multiple-cylinder, possessing 100-150 parts. Vickers estimated a flight weight prototype could be available (with appropriate funding) in 2 - 3 years based on a 1000 MTBF dosing objective.

SUNDSTRAND AVIATION, Denver, Colorado, October 18, 1963

Names of Participants

For Sundstrand: Dr. J. R. May, Chief Scientist; Tom Williams, Solar System Sales; John Landstrom, Chemical System Sales; Damon Phiney, Hydrazine APU (871A); George Kemp, Rubudium Solar Dynamic (ASTEC); Don Malohn, Biphenyl Solar Dynamic; Bob Loomis, Cryhocycle Systems.

Sundstrand had several dynamic engines whose characteristics make them potential lunar engines. They are:

1. Hydrazine dynamic
2. Integrated environmental systems
3. Solar dynamic closed systems

Hydrazine Power System

The hydrazine power system (871-A) is a high speed monopropellant expander for short duration missions. It has been developed to the point of flight rating tests. The problem areas are centralized on the decomposition of the N_2H_4 and the high turbine inlet temperature, $2000^{\circ}F$. This particular device was designed for over 10 kw net output, consisting of 115/200 volt, 400 cps electrical output and high pressure hydraulic power. System weight is

155 lbs, and its envelope is an eliptical cylinder (12 in. x 18 in. x 23 in.). The present fuel consumption is 5.09 lb/kw hr at the turbine shaft, and this is potentially 4.69 lb/kw hr.

Waste heat rejection from the system is negligible because of the temperature level. The exhaust cannot be retained for regeneration in any convenient manner. All of the useful power output of the system must eventually be rejected to a heat sink external to the system, because the propellant has little heat sink capacity.

Startup can be accomplished by injecting N_2O_4 with the hydrazine or by electrically heating the propellant. Voltage during transient loads can be held to ± 2.5% within 0.5 seconds. The exhaust would consist of hot N_2 and H_2.

The system has been designed for flight conditions so it can withstand vibration as well as thermal shock similar to that encountered on startup. The preliminary MTBF without redundancy is 1500 hrs (est.).

Integrated EC-Power System

Integrated systems include both the series-integrated H_2O_2 and the cryhocycle.

A cryhocycle with a turbine expander will have a slightly higher propellant consumption than the piston expander over the 1-10 kw power range. The model 1430 will be considered as a typical cryhocycle. It has 2 stage piston expansion with a 2 stage compressor. This particular cryhocycle is still in the design stage and is not scheduled for further development; however, the cryhocycle Turbine is ready for testing. Bearing problems have delayed testing.

The output is 400 cps 115/200 volt and 28 volt dc with voltage regulation of ± 1.5 percent. This 3 kw system weighs 46 lbs and fits a space 11 x 24.7 x 9.5 inches. Propellant consumption is around 1.5 lb/kw hr at the output shaft. Because of the low operating temperatures, there is no heat rejection from the prime mover.

Retention of the cryhocycle exhaust (hydrogen) is difficult. A fully integrated cryhocycle will have a cooling capacity equal to the heat load. Although it might be possible to start the prime mover on pressurized gas, motoring is considered desirable. Overloads can be handled with ± 3 percent voltage fluctuation and a slight increase in the propellant consumption.

This system was designed for use in a space environment and can tolerate moderate temperature fluctuations. The design MTBF is 2000 hrs, but no testing has been done to verify this number.

The X-20 System

This is a series integrated thermal control and power system. Cryogenic Hydrogen is expanded from -420°F to 200°F by exchanging with the coolant system for electronic equipment and environmental control. The hydrogen then enters a combustion chamber where oxygen is introduced and a gas weight ratio if 1 lb H_2:1 lb O_2 is exhausted at 1500°F. This gas drives a 3 stage turbine. A gear box is hermetically connected to the turbine shaft and provides shaft power to an electrical alternator and/or hydraulic pumps. Turbine exhaust gases are fed back to the thermal heat exchanger to preheat the cryogenic hydrogen and also dumped to space. At the present time this is not designed as a condensable system, and is a prolific user of hydrogen. The system could be designed with a product recovery unit.

For the open cycle system, and SPC of 2.27 lbs/kw hr at the gear box output shaft is anticipated. Alternator and hydraulic pump losses would increase this SPC and cannot be stated unless the mission is stipulated. Sundstrand believes the system will exhibit a very flat SPC characteristic over a 9 to 22.5 kw power range.

Some problems that remain are: 1. Freezing of the coolant by cryogenic hydrogen in the thermal control exchanger; 2. Control of turbine inlet temperature; 3. Life of hermetic seal; 4. Start up Procedure; and 5. Speed control.

Astronuclear

Solar Dynamic Systems

There are two solar dynamic systems under development at Sundstrand, the Biphenyl Rankine and the Rubidium Rankine.

Biphenyl Rankine

The solar powered Biphenyl Rankine system has been given preliminary considerations in the 1-25 kw range for space power systems. Accomplishment of the required reliability would take from 2-7 years, depending on the degree of redundancy. Materials considerations would be one of the more important efforts.

A proposed 10 kwe system modified for use on the moon would weigh about 900 lb. The comparable heat rejection at 300°F would be 47 kwt plus rejection of the used output. The system must be heated to about 500°F before mechanical starting is permitted. This preheating is accomplished by circulating heated helium through all critical components. After reaching the desired preheat temperature, the system can be started by valve controls, assuming correct orientation has been achieved. Total startup energy is between 700-1000 watt hrs.

The temperature fluctuations considered during storage of the system permit freezing of the working fluid thus making it storeable over a very wide range of temperature. The effect of solar radiation and lunar dust on the solar collector are uncertain at this time. Design life will be around 8000 hrs.

Rubidium Rankine

The ASTEC Solar power system is a Rubidium Rankine cycle capable of 15 kwe. It is in the hardware procurement stage as well as a materials/temperature limitation study. The Rubidium Rankine cycle is considered applicable in 5-25 kw range.

For the 15 kwe size the approximate specific weight is 90 lb/kw. Waste heat rejection will amount to 19,000 btu/kw hr.

The "boiler" is a double shell system with NaF in the inner shell receiving the direct energy from the collector and heated to approximately 1800°F. The outer shell contains LiH which is also heated with energy from the collector but the energy input is regulated to maintain the LiH near its melting point of approximately 1250°F. Heat of vaporization is supplied by the LiH and superheat and reheat is supplied by the NaF.

Startup will require preheating of about 4000 watt-hrs. Actual mechanical starting could be accomplished by expulsion of the working fluid into the cycle from a storage reservoir.

The solar collector and radiators are vulnerable to the environment (radiation, meteoroids and dust). Considering all known factors, the life expectancy will be of the order of 10,000 hrs.

BATTERY INFORMATION

The following trips were made to discuss present and projected performance of hermetically-sealed batteries.

1. NASA, Washington Office, 9/23 -- E. Cohn
2. National Bureau of Standards, Washington, 9/23 -- W. J. Hamer
3. Naval Research Laboratories, Washington, 9/24 -- J. C. White, C. P. Wales
4. Bell Telephone Laboratories, Murray Hill, 10/4 -- U. B. Thomas, D. R. Turner
5. USAERDL, Fort Monmouth, 10/14 -- D. Linden, A. Legath, H. Mandel
6. NASA Goddard, Greenbelt, 10/15 -- T. Hennigan, E. Stroup
7. Yardney Electric Corp., New York, 10/16 -- P. L. Howard.

In addition, two discussions were held at the New York meeting of the Electrochemical Society, 9/30 - 10/3, with

8. T. R. Beck, The Boeing Co. (Seattle)
9. A. Fleischer, Consultant (Orange, N.J.).

Astronuclear

The information obtained to date is summarized below.

There are only three secondary systems now undergoing vigorous development as hermetically-sealed storage batteries: nickel-cadmium (Ni-Cd), silver-cadmium (Ag-Cd) and silver-zinc (Ag-Zn). The lead-acid system involves formidable problems in sealing, in increasing life and in reducing the weight, so that there is as yet little interest in developing this system for space use (Thomas).

Primary batteries remain an attractive substitute for secondary batteries for such components as small portable communication modules, where the total energy requirement of the module is relatively small over its lifetime (1 to 5 years).

Primary Batteries
In terms of watt hours per pound output, sealed primary batteries now yield several times the output of sealed storage batteries. This relative performance is expected to continue into the future as lighter batteries of both types are developed, although the output ratio will be reduced to 2-3 eventually. The best primary system at present is the Ag-Zn, which yields 90-100 wh/lb. It has good charge retention, with less than 15 percent loss in capacity after a one-year stand at room temperature (Howard).

Current developments in magnesium batteries indicate that these may out-perform Ag-Zn in 1968. For example, the magnesium-dinitrobenzene battery is expected to yield 125-150 wh/lb (Linden). Present data indicate that the charge retention of magnesium batteries will be even better than that of Ag-Zn.

It may be concluded that such high energy densities will result in widespread use of primary batteries in self-powered solid state devices.

Nickel-Cadmium
Hermetically-sealed Ni-Cd batteries have undergone major developments in the last three years. Energy output of 15 wh/lb (initial full discharge) has

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been achieved, the same as for vented cells. The useful temperature range has been extended to above 50°C by substituting nylon or other synthetic separators for the older cellulosic separators. At these elevated temperatures, however, the energy output falls off by about 50 percent and the batteries have poor charge retention.

Storage at low temperatures, -50 to 0°C, is not detrimental and in fact improves charge retention (Thomas). At normal discharge rates the available energy is decreased at low temperatures, by more than a factor of 2 at -25°C. At room temperature rapid discharge (as fast as 30 min full discharge) can be tolerated without large losses in ampere-hour output.

The best cycling performance obtained to date indicates that a constant depth of discharge, D, of 60-75 percent per cycle can be maintained for 1000 cycles in the optimum temperature range 15 to 30°C (References 1, 2 and 3). At a smaller depth of discharge the cycle life C increases, and a logarithmic relationship is apparently obeyed:

$$\log C = a - bD.$$

The values of the constants a and b vary from one battery to another. However the total discharge energy over the lifetime of the battery increases as D decreases. Thus a life of 10,000 cycles has been obtained at 20-30 percent depth of discharge.

A numerical statement of present Ni-Cd performance can not be given concisely. Adequate standards of reliability have not been established as yet. Thus one measure of the life of a battery or group of cells has been taken to be the number of cycles at which one half of the cells failed (Reference 3) -- not a high-reliability end point. In addition, a given depth of discharge may refer to different wh/lb outputs, depending upon the size and make of cells incorporated in the battery.

Similarly a concise statement of predicted performance of Ni-Cd batteries in 1968 or 1972 can only be a rough guess. The consensus of opinion obtained

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from the specialists interviewed to date leads to an expected maximum performance for Ni-Cd in 1968 of 18 wh/lb and in 1972 of 20 wh/lb. For a life of 1000 cycles or more the output will be limited to about 80 percent of these values, namely 14 wh/lb in 1968 and 16 wh/lb in 1972.

Because of these relatively low outputs, it is probable that Ni-Cd batteries will be superseded by the silver batteries discussed below, in terms of space applications in 1968 and beyond.

Silver-Cadmium

Present Ag-Cd batteries have achieved double the wh/lb output of Ni-Cd in vented cells. In hermetically-sealed cells tested to date, however, the two systems yield about the same energy density. Lighter Ag-Cd batteries are now under rapid development (Hennigan). Considerable progress has been made in understanding gas buildup and failure of Ag-Cd batteries within the past two years, so that increasingly lighter battery designs will appear in quick succession as lighter casing, improved separator combinations and perhaps third electrodes are brought into play. Pressure buildup and other minor causes of failure can now be eliminated by proper design, so that battery life will be limited mainly by the rate of silver penetration of the separators used (Thomas). Since this factor is subject to control, it will be possible very soon to design Ag-Cd batteries to meet various specified space goals for cycle life.

Silver-cadmium cells suffer some disadvantages in comparison with Ni-Cd, but this is more than compensated by certain advantageous properties and by the inherently greater energy density.

1. The operating temperature limit for Ag-Cd is probably about 50°C, because of the use of cellulosic materials as part of the multicomponent separator. At higher temperature, cell capacity decreases irreversibly and relatively rapidly. On the other hand, Ni-Cd cells can recover from short term operation at 50-60°C without loss in capacity. At low temper-

atures, -25 to 0°C, Ag-Cd yields a somewhat higher percentage of its maximum capacity than does Ni-Cd, but both systems suffer a large percentage loss in cycle life (Reference 3).

2. Overcharging can be avoided almost entirely in Ag-Cd, the end-of-charge voltage inflection being used to terminate charging. For Ni-Cd, some overcharging is necessary to regain full output capability. As a result, a higher ampere-hour efficiency (97 percent) for the charge-discharge cycle can be obtained with Ag-Cd (Reference 4).

3. For maximum output Ag-Cd cannot be charged as rapidly as Ni-Cd, and even with projected improvements the rate will be limited to C/5 (Howard).

4. The charge retention of Ag-Cd is better. For fully charged cells stored in the temperature range -25 to 0°C, energy losses as low as 10 percent per year are anticipated (Howard).

The best cycling performance of sealed Ag-Cd batteries appears to be that obtained at Boeing (Reference 5). These data show that 1000 cycles can be obtained at a depth of discharge of about 80 percent. Over 13,000 cycles have been obtained to date at D = 25 percent. Present indication is that Ag-Cd may provide the best cycle life of any secondary system. Because of the paucity of long-term cycling data, however, such an indication is only a rough guide and is subject to considerable change before 1968.

The best composite estimate of future Ag-Cd battery performance, from the information obtained so far, may be given in terms of 1000-cycle life as 24 wh/lb in 1968 and 36 wh/lb by 1972. These estimates, together with the expectation of high reliability for Ag-Cd batteries, point to Ag-Cd supplanting Ni-Cd in space by 1968.

Silver-Zinc

Although the Ag-Zn system in gasket-sealed primary cells has exceeded 90 wh/lb, its output in storage batteries is much less. Irreversible loss of the active material from the negative plates, and penetration of separators by both zinc and silver metal, tend to shorten cycle life considerably. Hence

the batteries are provided with excessive quantities of negative electrode,
separator and electrolyte. Even with the extra weight in current cells, only
a small number of cycles have been obtained at depths of discharge greater
than 50 percent (e.g., 11-13 cycles at D = 75 percent, 25-35°C, Reference 3).
More encouraging results have been reported at D = 25% (Reference 6),
where batteries or groups of cells cycled at 50-100°F yielded 400 cycles be-
fore the first cell of each group failed. The latter results were achieved
with sealed cells of 33 wh/lb initial capacity which were potted in epoxy
resin. They indicate that rapid improvements are being made in hermetic
seals and in cycle life.

Unsealed low-rate Ag-Zn storage batteries have been made with excellent
charge retention (about 10 percent loss per year), with good output charac-
teristics at fairly low temperatures (somewhat better than Ag-Cd), and with
approximately the same charging rate limitations and high temperature lim-
itations as Ag-Cd (Howard). These batteries have attained initial capacities
of 45-50 wh/lb, but a considerable reduction in capacity will accompany cur-
rent modifications aimed at reliable sealed cells.

The best judgement from the information received to date is that the Ag-Zn
storage battery will advance only to about 24 wh/lb by 1968 for a life of 1000
cycles. This is the same energy density as predicted above for Ag-Cd. And
since the latter is expected to be more reliable in its performance from cell
to cell, Ag-Zn secondary batteries will probably have little use in space in
1968 except for missions involving considerably less than 1000 cycles.

By 1972, however, this situation should reverse. By then, a 1000-cycle life
is expected to be attained at about 48 wh/lb. This is one-third higher than
the final value of 36 wh/lb estimated for Ag-Cd.

In conclusion, it is predicted that the Ni-Cd battery will be essentially ob-
solete by the time the lunar engine-fuel system is finalized in 1968. Silver-
cadmium will be the main storage battery system. The higher energy Ag-Zn

or magnesium batteries will be employed where small primary cells can be substituted. In addition, Ag-Zn storage batteries will be considered for missions involving a relatively small number of cycles. By 1972 the Ag-Zn battery is expected to displace Ag-Cd as the basic secondary system.

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Astronuclear

B. CALCULATION OF OPTIMUM RADIATOR WEIGHT
FOR HEAT TRANSFER ON THE LUNAR SURFACE

This analysis assumes:

1. Constant fin thickness

2. External radiation incident on fin is related to an effective

   space temperature $T_s{}^*$ based on radiator geometry and coating

   properties.

3. Radiation is from one surface of the fin, with the back

   insulated.

4. Infrared emissivity and solar reflectivity remain constant

   and are:

   ϵ_r = .85 (A value of α_s = .10 would represent a very

   α_s = .10 optimistic case and one possibly attainable

   ϵ_s = 1.0 in the future.)

5. The temperature gradient along the width of the fin is small and that this variation can be approximated by a mean temperature T_o at the midpoint of the width.

6. The radiator geometry is a tent-like shape placed 45° to the vertical.

7. Aluminum as both fin and tube material.

8. A heat transfer fluid with a specific gravity of 1.0.

9. A fixed fin thickness of .015 inch.

10. The source temperature is equal to the mean fin root temperature, T_o.

The equations used are:

$$T_o = \left(\frac{T \text{ in}^4 - T \text{ out}^4}{2} \right)^{1/4}$$

$$T_s^* = \left(\frac{Q \text{ solar} + Q \text{ lunar} + Q \text{ albedo}}{\sigma \epsilon_s} \right)^{1/4}$$

$$w = \frac{q \text{ in.}}{632.5 \sqrt{t} \sqrt{Kt_r (10^9) \left(\frac{T_o}{1000} \right)^5}} \cdot \sqrt{f(L)} \quad \text{-ft}$$

where $\quad f(L) = 1 - \left(\frac{T_e}{T_o} \right)^5 - 5 \left(\frac{T_s^*}{T_o} \right)^4 \left(1 - \frac{T_e}{T_o} \right)$

$$1 = \frac{L}{\sqrt{\frac{\rho E(10^9)}{Kt}} \left(\frac{T_o}{1000} \right)^3} \quad \text{- ft}$$

Where L is point at which $\left| \frac{dT}{dX} \right| = 0$

Assembly weight:

$$W_{total} = W_{fin} + W_{tube} + W_{fluid}$$

$$W_{fin} = l \cdot t \cdot w \cdot \rho_{fin}$$

$$W_{tube} = \frac{\pi (D_o^2 - D_i^2) \cdot w \cdot \rho_{tube}}{4}$$

$$W_{fluid} = \frac{\pi D_i^2 \cdot w \cdot \rho_{fluid}}{4}$$

In this analysis, an effective space temperature is used to take into effect the non-zero sink temperature conditions.

It is defined as

$$T_s^{*4} = \frac{Q_T'}{\sigma \epsilon}$$

Where

Q_T' is total heat flux $\frac{BTU}{HR \ FT^2}$ into radiator due to solar radiation, lunar radiation and albedo. This quantity has been calculated for various radiator geometry as a function of sun phase angle at the lunar equator.

For this analysis, the geometry chosen was a tent-like radiator placed at an angle of 45° to the vertical $\alpha_s = .1$ $\epsilon_R = .90$

For the worse case; ie $\phi = 0°$

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$$Q_T^! = 82.5 \quad \frac{BTU}{HR \ FT^2} \quad , \ \epsilon = 1.0$$

$$T_s^* = \left(\frac{82.5}{.0.1713 \times 10^{-8}} \right)^{1/4} = (482 \times 10^8)^{1/4}$$

$$T_s^* = 470°R = (10°F)$$

The procedure for obtaining $Q_T^!$ is as follows:

Heat flux onput to radiator placed at 45° to vertical @ lunar

equator

back side insulated

Q input = Q solar + Q lunar + Q albedo

Area = dA sin θ

solar incidence $= Q_s$ $\sin \theta \cos \phi + \cos \theta \sin \phi$

@ $\phi = 0$ $Q = Q_s (\sin \theta)$

C $\phi = 90°$ $Q = Q_s (\cos \theta)$

C $\phi = \theta$ $Q = Q_s (\sin^2\theta + \cos^2\theta) = Q_s$

For particular case

$\theta = 45°$

$\sin \theta = \cos \theta = .707$, $Q_s = 430 \dfrac{BTU}{HR\ FT^2}$

$Q'\,solar = Q_s\,\alpha_s\,(.707)\ \cos \phi + \sin \phi$

$Q'\,solar = 430 \times 0.1 \times .707\ \cos \phi + \sin \phi$

$Q'\,solar = 30.4\ \cos \phi + \sin \phi$

ϕ	$\cos \phi$	$\sin \phi$	$30.4\ \cos \phi + \sin \phi$
0	1.0	0	30.4
30	.866	.50	41.5
60	0.500	.866	41.5
90	0	1.0	30.4
270	0	1.0	30.4
300	.500	.86	41.5
330	.866	.50	41.5
360	1.0	0	30.4

Lunar infrared $= \sigma \, t_L \, F_{LR} \, T_L^4 \, t_R$

$\qquad F_{LR} = 1 \, 2 \, (1 + \cos \beta)$

$\qquad$ For $\theta = 45°$

$\qquad\qquad \beta = 180 - \theta$

$\qquad \cos (180 - \theta) = \cos (\theta) = \cos 45° = -.707$

$\qquad F_{LR} = \dfrac{1}{2} \, (1 - 607) = \dfrac{.293}{2} = .147$

Lunar $= (.0.1713 \times 10^{-8} \times .9 \times .147 \times .9) \, T_L^4$

Q lunar $= .0204 \times 10^{-8} \, T_L^4$

$\qquad$ Q' lunar $= .0204 \quad 10^{-8} \quad T_L^4$

ϕ	$T_L \, °R$	T_L^4	$.0204 \times 10^{-8} \, T_L^4$
0	701	2400×10^8	49.2
30	670	1980×10^8	40.5
60	600	1290×10^8	26.4
90	350	717×10^8	14.7
120	250	390×10^8	8.0
150	210	193×10^8	3.95
180	190	129×10^8	2.64
210	170	82.5×10^8	1.69
240	165	73.2×10^8	1.50
210	162	68.5×10^8	1.4
300	600	1290×10^8	26.4
330	670	1980×10^8	40.5
360	701	2400×10^8	49.2

Q REFL = (430 cos ϕ) (albedo) (α_s) (F_{LR})

Q REFL = 430 cos ϕ x .124 x .10 x .147 = 0.79 cos ϕ

Q'REFL = 0.79 cos ϕ

ϕ	cos ϕ	2.95 cos ϕ
0	1.0	.79
30	0.866	.685
60	0.500	.40
90	0	0
270	0	0
300	0.500	.40
330	0.866	.685
360	1.0	.79

ϕ	Q solar	Q lunar	Q REFL.	Q total $\frac{BTU}{HR\ FT^2}$
0	30.4	49.2	0.79	80.4
30	41.5	40.5	0.685	82.7
60	41.5	26.4	0.40	68.3
90	30.4	14.7	0	45.1
120	0	8.0	0	8.0
150	0	3.95	0	3.95
180	0	2.64	0	2.64
210	0	1.69	0	1.70
240	0	1.50	0	1.5
270	30.4	1.40	0	31.8
300	41.5	26.4	0.40	68.3
330	41.5	40.5	0.685	82.7
360	30.4	49.2	0.79	80.4

Ideally, if T_{ie} = 60°F = 520°R   ←—— assume.

$$\text{and } T_{io} = 120 \quad = \quad 580°R$$
$$150 \quad = \quad 610°R$$
$$300°F = \quad 760°R$$
$$600°F = \quad 1060°R$$
$$800°F = \quad 1260°R$$

then

$$T_o = \left(\frac{T_{io}^4 + T_{iw}^4}{2} \right)^{1/4}$$

T_{io}	T_{io}^4	T_{iw}	T_{iw}^4	$(T_{io}^4 + T_{iw}^4)2$	T_o
°R	x 10^8	°R	x 10	x 10^8	°R
580	1,130	520	730	930	552
610	1,380	520	730	1,055	571
760	3,350	520	730	2,040	674
1060	12,500	520	730	6,615	905
1260	25,100	520	730	12,915	1068

$$
\begin{aligned}
\text{if} \quad T_{2m} &= 40°F = 500°R \\
60 &= 510°R \\
100 &= 560°R \\
200 &= 660°R \\
400 &= 860°R \\
600 &= 1060°R
\end{aligned}
$$

$$T_{im} - T_{2m} = \begin{matrix} 552 - 550 \\ 570 - 510 \\ 674 - 560 \\ 405 - 660 \\ 1068 - 860 \end{matrix} = \begin{matrix} 2°R \\ 60°R \\ 114°R \\ 245 \\ 208 \end{matrix} \qquad (T_{im} - T_{2m}) = \begin{matrix} 2°R \\ 10 \\ 50 \\ 100 \\ 200 \\ 300 \end{matrix}$$

			$(T_o - T_e)$			
$T_{1m} - T_{2m}$	2	10	50	100	200	300
T_{1m} °R	T_e	T_e	T_e	T_e	T_e	T_e
552	550	542	502	—	—	—
571	569	561	521	471	—	—
674	672	664	624	574	474	—
905	903	895	855	805	705	605
1068	1066	1058	1018	968	868	768

$$T_s^* = 470°R$$

T_o °R	$\left(\dfrac{T_o}{1000}\right)^3$	$\left(\dfrac{T_o}{1000}\right)^4$	$\left(\dfrac{T_s^*}{T_o}\right)$	$(T_o - T_s^*)$ °R	$\left(\dfrac{T_o}{1000}\right)^5$	$\left(\dfrac{T_s^*}{T_o}\right)^5$
552	.177	.0930	.85	82	.0513	.44
571	.186	.1055	.822	101	.0602	.374
674	.305	.2040	.698	204	.1372	.166
905	.819	.6615	.52	435	.5980	.0389
1068	1.21	1.290	.44	598	1.378	.0166

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$(T_o - T_e)$			$T_s^* = 470°R$			
T_o		2		10		50
°R	T_e	$T_e - T_s^*$	T_e	$T_e - T_s^*$	T_e	$T_e - T_s^*$
552	550	80	542	72	502	32
571	569	99	561	91	521	51
674	672	202	664	194	624	154
905	903	433	895	425	855	385
1068	1066	596	1058	588	1018	548

$(T_o - T_e)$			$T_s^* = 470°R$			
T_o		100		200		300
°R	T_e	$T_e - T_s^*$	T_e	$T_e - T_s^*$	T_e	$T_e - T_s^*$
552	452	—	352	—	252	—
571	471	1	371	—	271	—
674	574	104	474	4	374	—
905	805	335	705	235	605	135
1068	968	498	868	398	768	298

T_o °R	$\frac{T_s^*}{T_o}$	(T_o-T_e)	$T_s^* = 470°R$	
		2	10	50
		$(T_e-T_s^*/T_o-T_s^*$	$(T_e-T_s^*/T_o-T_s^*$	$(T_e-T_s^*/T_o-T_s^*$
552	.85	.975	.878	.39
571	.822	.980	.90	.504
674	.698	.991	.95	.7
905	.52	.995	.978	.886
1068	.44	.997	.984	.916

T_o °R	$\frac{T_s^*}{T_o}$	100	200	300
		$(T_e-T_s^*/T_o-T_s^*$	$(T_e-T_s^*/T_o-T_s^*$	$(T_e-T_s^*/T_o-T_s^*$
522	.85	——	——	——
571	.822	.099	——	——
674	.698	.51	.196	——
905	.52	.77	.54	.31
1068	.44	.833	.666	.498

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$(T_o - T_e)$ $T_s^* = 470°R$

T_o °R	$\frac{T_s^*}{T_o}$	2		10		50	
		$\frac{(T_e-T_s^*)}{(T_o-T_s^*)}$	L	$\frac{(T_e-T_s^*)}{(T_o-T_s^*)}$	L	$\frac{(T_e-T_s^*)}{(T_o-T_s^*)}$	L
552	.85	.975	.15	.878	.30	.39	.95
571	.822	.980	.12	.90	.25	.504	.79
674	.698	.991	.05	.95	.20	.70	.60
905	.52	.995	.02	.978	.15	.886	.40
1068	.44	.997	.01	.984	.11	.916	.35

$(T_o - T_e)$ $T_s^* = 470°R$

T_o °R	$\frac{T_s^*}{T_o}$	100		200		300	
		$\frac{(T_e-T_s^*)}{(T_o-T_s^*)}$	L	$\frac{(T_e-T_s^*)}{(T_o-T_s^*)}$	L	$\frac{(T_e-T_s^*)}{(T_o-T_s^*)}$	L
552	.85	—	—	—	—	—	—
571	.822	.099	1.95	—	—	—	—
674	.698	.51	.90	.196	1.77	—	—
905	.52	.77	.62	.54	1.15	.31	1.92
1068	.44	.833	.50	.666	.87	.498	1.32

$$1 = \frac{L}{\sqrt{\dfrac{\sigma E\ (10^9)}{Kt}\left(\dfrac{T_o}{1000}\right)^3}}$$

Where $\quad \sigma = 0.1713 \times 10^{-8} \quad \dfrac{BTU}{HR\ FT^2\ F}$

$E = \epsilon_t = 0.85$

$K = 120 \quad \dfrac{BTU}{HR\ FT^2\ F} \qquad$ aluminum

$t = .015\ in = .00125\ ft$

$$\frac{\sigma E\ (10^9)}{Kt} = \frac{0.713 \times 10^{-8} \times .85 \times 10^9}{120 \times .00125} = .00097 \times 10^4$$

$$\frac{\sigma E\ (10^9)}{Kt} = 9.7 \quad \frac{1}{FT^2 {}^\circ R^3}$$

$T_o\ {}^\circ R$	$\left(\dfrac{T_o}{1000}\right)^3$	$\dfrac{\sigma E\ (10^9)}{Kt} \cdot \left(\dfrac{T_o}{1000}\right)^3 \dfrac{1}{ft^2}$	$\sqrt{\dfrac{\sigma E\ (10^9)}{Kt} \cdot \left(\dfrac{T_o}{1000}\right)^3} \dfrac{1}{ft}$
552	.177	1.715	1.31
571	.186	1.80	1.341
674	.305	2.96	1.721
905	.819	7.92	2.82
1068	1.21	11.72	3.42

T_o		(T_o-T_e)		$T_s{}^* = 470°R$		
	2		10		50	
°R	L	1 ft	L	1 ft	L	1 ft
552	.15	.0725	.30	.219	.95	.725
571	.12	.085	.25	.187	.79	.587
674	.05	.029	.20	.116	.60	.348
905	.02	.0071	.15	.053	.40	.142
1068	.01	.00292	.11	.0322	.35	.103

T_o		(T_o-T_e)		$T_s{}^* = 470°R$		
	100		200		300	
°R	L	1 ft	L	1 ft	L	1 ft
552	—	—	—	—	—	—
571	1.95	1.45	—	—	—	—
674	0.90	.522	1.77	1.03	—	—
905	0.62	.220	1.15	0.407	1.92	0.68
1068	0.50	.146	0.87	0.255	1.32	0.386

Calculation of Fin width for a heat onput of

1 kw (3,417 BTU/HR)

$$W = \frac{Q_{in}}{632.5 \sqrt{t \quad EK\sigma(10^9)\left(\frac{T_o}{1000}\right)^5 \cdot f(L)}}$$

$$= 6.32.5 \quad t \sqrt{EK\sigma(10^9)} \quad \frac{T_o}{1000}^{5}$$

$$= 632.5 \left(EK\sigma(10^9)\cdot t\right)^{1/2} \quad \frac{T_o}{1000}^{5}$$

$$= 632.5 \left(.85 \times 10 \times 0.1713 \times 10^{-8} \times 10^9 \times .015\right)^{1/2} \quad \frac{T_o}{1000}^{5}$$

$$F(L) = 1 - \frac{T_e}{T_o}^5 - 5\frac{T_s^*}{T_o}^5 \quad 1 - \frac{T_e}{T_o}$$

$(T_o - T_e)$

T_o °R	$\frac{T_o}{1000}$ 5	2		10		50	
		L	f(L)	L	f(L)	L	f(L)
552	.0513	.15	.0112	.30	.0396	.95	.141
571	.0602	.12	.0107	.25	.0435	.79	.172
674	.1372	.05	.041	.20	.045	.60	.234
905	.5980	.02	.0092	.15	.0352	.40	.225
1068	1.378	.01	.0096	.11	.0423	.35	.201

$(T_o - T_e)$

T_o °R	$\frac{T_o}{1000}$ 5	100		200		300	
		L	f(L)	L	f(L)	L	f(L)
552	.0513	—	—	—	—	—	—
571	.0602	1.95	.26	—	—	—	—
674	.1372	0.90	.385	1.77	.46	—	—
905	.5980	0.62	.399	1.15	.65	1.92	.775
1068	1.378	0.50	.371	0.87	.610	1.32	.74

T_o °R	2		10		50	
	$\dfrac{T_e}{T_o}$	$\left(\dfrac{T_e}{T_o}\right)^5$	$\dfrac{T_e}{T_o}$	$\left(\dfrac{T_e}{T_o}\right)^5$	$\dfrac{T_e}{T_o}$	$\left(\dfrac{T_e}{T_o}\right)^5$
552	.995	.9758	.982	.9132	.91	.624
571	.995	.9758	.983	.9179	.912	.628
674	.998	.9900	.985	.9272	.925	.677
905	.998	.9900	.99	.951	.945	.754
1068	.998	.9900	.991	.956	.954	.790

T_o °R	100		200		300	
	$\dfrac{T_e}{T_o}$	$\left(\dfrac{T_e}{T_o}\right)^5$	$\dfrac{T_e}{T_o}$	$\left(\dfrac{T_e}{T_o}\right)^5$	$\dfrac{T_e}{T_o}$	$\left(\dfrac{T_e}{T_o}\right)^5$
552	—	—	—	—	—	—
571	—	—	—	—	—	—
674	—	—	—	—	—	—
905	.89	.559	—	—	—	—
1068	.906	.6105	.813	.355	—	—

T_o	$\left(\dfrac{T_s*}{T_o}\right)^4$	2		10		50	
		$1-\left(\dfrac{T_e}{T_o}\right)^5$	$1-\dfrac{T_e}{T_o}$	$1-\left(\dfrac{T_e}{T_o}\right)^5$	$1-\dfrac{T_e}{T_o}$	$1-\left(\dfrac{T_e}{T_o}\right)^5$	$1-\dfrac{T_e}{T_o}$
552	.522	.0242	.005	.0868	.018	.376	.09
571	.454	.0242	.005	.0821	.017	.372	.088
674	.237	.0100	.002	.0628	.015	.323	.075
905	.076	.0100	.002	.049	.010	.246	.055
1068	.0375	.0100	.002	.044	.009	.210	.046

T_o	$\left(\dfrac{T_s*}{T_o}\right)^4$	100		200		300	
		$1-\left(\dfrac{T_e}{T_o}\right)^5$	$1-\dfrac{T_e}{T_o}$	$1-\left(\dfrac{T_e}{T_o}\right)^5$	$1-\dfrac{T_e}{T_o}$	$1-\left(\dfrac{T_e}{T_o}\right)^5$	$1-\dfrac{T_e}{T_o}$
552	—	—	—	—	—	—	—
571	—	—	—	—	—	—	—
674	—	—	—	—	—	—	—
905	.441	.11	—	—	—	—	—
1068	.389	.094	.645	.187	—	—	—

$\left(\dfrac{T_s *}{T_o}\right)^4$	$5\left(\dfrac{T_s}{T_o}\right)^4$
.522	2.61
.454	2.265
.237	1.185
.076	..38
.0375	.188

$$\left] = 632.5\,(.218)^{1/2}\left(\left(\frac{T_o}{1000}\right)^5 = 296\left(\frac{T_o}{1000}\right)^5\right.\right.$$

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(T_o-T_e)

T_o °R	$\sqrt{\dfrac{T_o}{1000}}^5$	2		10		50	
		$f(L)$	$f(L)$	$f(L)$	$f(L)$	$f(L)$	$f(L)$
.552	.2265	.0112	.106	.0396	.199	.141	.376
.571	.246	.0107	.1035	.0435	.2085	.172	.415
674	.370	.041	.2025	.045	.212	.234	.484
905	.773	.0092	.096	.0352	.1875	.225	.475
1068	1.162	.0096	.098	.0423	.206	.201	.448

(T_o-T_e)

T_o °R	$\sqrt{\dfrac{T_o}{1000}}^5$	100		200		300	
		$f(L)$	$f(L)$	$f(L)$	$f(L)$	$f(L)$	$f(L)$
.552	.2265	—	—	—	—	—	—
.571	.246	.26	.510	—	—		
674	.370	.385	.620	.46	.679	—	—
905	.773	.399	.631	.65	.806	.775	.88
1068	1.162	.371	.610	.610	.781	.74	.86

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$$W = \frac{Q_{in}}{296 \left(\frac{T_o}{1000}\right)^5}$$

$Q_{in} = 1\ kw\ (3,471\ BTU/HR)$

T_o °R	$\dfrac{Q_{in}}{296\left(\frac{T_o}{1000}\right)^5}$	$\dfrac{(T_o - T_e)}{2}$ W ft	10 W ft	50 W ft	100 W ft	200 W ft	300 W ft
552	50.8	480	255	136	——	——	——
571	46.8	452	225	113	92	——	——
674	31.2	154	147	64.6	50.5	46	——
905	14.9	156	79.5	31.4	23.6	18.5	16.9
1068	9.92	101.5	48.1	22.1	16.3	12.7	11.6

Q in = 1 kw , t = .015 in

T_o	$(T_o - T_e)$					
	2		10		50	
°R	1 ft	W ft	1 ft	W ft	1 ft	W ft
552	.0725	480	.219	255	.725	136
571	.085	452	.187	225	.587	113
674	.029		.116	147	.348	64.6
905	.0071	156	.053	79.5	.142	31.4
1068	.00292	101.5	.0322	48.1	.103	22.1

Q in = 1 kw , t = .015 in

T_o	100		200		300	
°R	1 ft	W ft	1 ft	W ft	1 ft	W ft
552	___	___	___	___	___	___
571	1.45	92	___	___	___	___
674	0.522	50.5	1.03	46	___	___
905	0.220	23.6	0.407	18.5	.68	16.9
1068	0.146	16.3	0.255	12.7	.386	11.6

T_o	$(T_o - T_e)$		10		50	
°R	A ft^2	Wt lb	A	Wt	A	Wt
552	34.8	7.45	55.9	12.0	98.5	21.1
571	38.4	8.2	42.0	9.0	66.4	14.2
674	4.46	.955	17.1	3.66	22.6	4.84
905	1.110	.238	4.21	0.91	4.45	0.95
1068	.297	.0635	1.56	0.334	2.28	0.49

T_o	100		200		300	
°R	A	Wt	A	Wt	A	Wt
552	——	——	——	——	——	——
571	134	28.6	——	——	——	——
674	26.4	5.65	47.6	10.2	——	——
905	5.2	1.11	7.55	1.62	11.7	2.5
1068	2.38	.51	3.23	0.69	4.5	0.963

NOTE: Wt <u>does not</u> include Wt of supply tube and is for 1/2 of total fin area.

$$Wt = \rho \, Vol = \rho \cdot t \cdot w \cdot 1 =$$

$$\rho \, al = .099 \frac{lb}{in} \times 1728 \frac{in^3}{ft^3} = 171 \frac{lb}{ft^3}$$

$$\rho \cdot t \cdot = 171 \frac{lb}{ft^3} \times \frac{.015}{12} ft = 0.214 \frac{lb}{ft^2}$$

In using the foregoing table, cantron must be used. The weights listed are for the <u>fin</u> only. Therefore, the short fins appear to be the optimum design until the weight of the supply tubes is included. Since this value is a constant term for a given tube size and material, a wt./length of tube must be determined and added to the fin to obtain the overall weight per tube or kw/lb factor. The fluid weight will also be included, using a specific gravity of 1.0.

$$Tube \left(\frac{vol}{ft}\right) = \frac{\pi}{4} (D_o^2 - D_i^2) \times 1.4$$

$$tube \frac{Wt}{ft} = \rho \frac{\pi}{4} (D_o^2 - D_i^2)$$

$$tube \frac{Wt}{ft} = \frac{171 \, \pi}{4} \left(\frac{9}{64} - \frac{1}{16}\right) \times \frac{ft^2}{144 \, in^2} = \frac{171 \times \pi \times 5}{4 \times 64 \times 144}$$

$$tube \frac{Wt}{ft} = .073 \; lb/ft$$

fluid $\quad \dfrac{Wt}{ft} = 62.4\ \dfrac{lb}{ft^3} \times \dfrac{\pi}{4} D_i^2 = \dfrac{62.4 \times \pi}{4} \times \dfrac{1}{16} \times \dfrac{ft^2}{144^2}\quad 1\ ft$

fluid $\quad \dfrac{Wt}{ft} = \dfrac{62.4 \times \pi \times 1}{4 \times 16 \times 144} = .0212\ lb/ft$

Total $= .073 + .0212 = .0952 = .7\ \dfrac{lb}{ft}\quad$ of tube

This means that for every ft of supply tube a factor of $0.1\ \dfrac{lb}{ft}$ must be included. When using aluminum tubing (3/8 OD, 1/4 ID) and a fluid having a S.G. of 1.0.

$$W_{assembly} = W_{tube} + W_{fluid} + W_{fins}$$

![Astronuclear logo] Astronuclear

T_o-T_e	2		10		50	
To^5	W_{Fin} lb	W_{Total}	W_{Fin} lb	W_{Total}	W_{Fin}	W_{Total}
552	14.9	62.9	24	49.5	42.2	55.8
571	16.4	61.6	18	40.5	24.4	35.7
674	1.81	18.2	7.4	22.1	9.68	16.14
905	.48	16.1	1.82	9.87	1.90	5.04
1068	.128	10.33	.668	5.48	.98	3.19

	100		200		300	
To^5	W_{Fin}	W_{Total}	W_{Fin}	W_{Total}	W_{Fin}	W_{Total}
552	—	—	—	—	—	—
571	57.2	66.4	—	—	—	—
674	11.3	16.4	20.4	25.0	—	—
905	2.22	4.59	3.24	5.09	5.0	6.69
1068	1.02	2.65	1.38	2.65	1.94	3.10

	$T_o - T_e$		2		10		50	
R		oR	$\frac{Q}{lb}$ $\frac{KW}{lg}$					
To	Te			Te	$\frac{Q}{lb}$	Te	$\frac{Q}{lb}$	
552	550		.0318	542	.0406	502	.0358	
571	569		.0324	561	.0495	521	.056	
674	672		0.110	664	.0905	624	.124	
905	903		0.124	895	.203	855	.397	
1068	1066		0.193	1058	.365	1018	.628	

R	100		200		300	
To	Te	$\frac{Q}{lb}$	Te	$\frac{Q}{lb}$	Te	$\frac{Q}{lb}$
552	—	—	—	—	—	—
571	471	.0302	—	—	—	—
674	574	.122	474	.0802	—	—
905	805	.436	705	.394	605	.30
1068	968	.755	868	.756	768	.646

![Astronuclear logo]

From Radiator Optimization Plot

T_o °R	Max. $\frac{kw}{lb}$	Fin Length (ft)	Fin Width (ft)	Fin Efficiency η
552	.041	.70	140	.51
571	.056	.587	113	.59
674	0.125	.420	51.0	.64
905	0.44	.30	21	.60
1068	0.775	.26	12.7	.58

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CLADDING CALCULATIONS

Determine:

(a) Cladding thk. for .99 prob. & .999 prob. use one segment per 1 kw.

(b) Weight for both cases with and without coolant

Avg. Source Temp. °R	Fin width per kw (ft)	Exposed Area ft^2
552	140	4.37
600	90	2.81
700	44	1.372
900	20	0.625
1100	14	0.437

Exposed area = projected area of tube x width projected area of tube =

Dia. x W

$$\text{Area} = \frac{D_o}{12} \cdot W \text{ ft}^2 = \frac{\frac{3}{8}}{12} \cdot W = \frac{W}{32} \text{ ft}^2$$

For 3 year mission

3 years = 1090 days

Since Exposure = $\frac{\text{Area x Time}}{1 - P_o}$

For $P_o = .99$

$$\text{Exposure} = \frac{1090 \times \text{Area}}{.01} = 1.09 \times 10^5 \text{ Area} \longrightarrow P_o = .99$$

$$= 1.09 \times 10^6 \text{ Area} \longrightarrow P_o = .999$$

1 kw Per Segment

Avg. Source Temp. °R	Exposed Area ft^2	Exposure ft^2-day		Thickness in	
		$P_o = .99$	$P_o = .999$	$P_o = .99$	$P_o = .999$
552	4.37	4.77×10^5	4.77×10^6	0.21	0.43
600	2.81	3.07×10^5	3.07×10^6	0.18	0.39
700	1.372	1.495×10^5	1.495×10^6	0.14	0.305
900	0.625	0.680×10^5	$.680 \times 10^6$	0.11	0.235
1100	0.437	0.476×10^5	0.476×10^6	0.10	0.210

Weight of Clad per foot

$$\frac{Wt}{ft} = \dot{\rho} \; vol. = \rho \; \frac{\pi \, D_o^2}{4} - \frac{\pi \, D_i^2}{4} \times \frac{1}{144} = \frac{\pi \rho}{4} \, (D_o^2 - D_i^2) \times \frac{1}{144}$$

$$D_o = (D_i + 2h)$$

$$(D_o^2 - D_i^2) = 4h \, D_i + 4h^2 = (rh \, D_i + 4h^2)$$

$$D_i = 3/8$$

$$\frac{Wt}{ft} = \frac{171 \, \frac{lb}{ft^3} - \pi}{4 \times 144 \, \frac{in^2}{ft2}} \quad (4h \, D_i + 4h^2) = 3.74 \; (h \, D_i + h^2$$

$$\frac{Wt}{ft} = 3.74 \quad (.375 \, h + h^2) \; \frac{lb}{ft} = 3.74 \, h \, (.375 + h)$$

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Avg. Source Temp. °R	Thickness h		h(.375 th)		Wt/ft	10/ft
	$P_o=.99$	$P_o=.999$	$P_o=.99$	$P_o=.999$	$P_o=.99$	$P_o=.999$
552	0.21	0.43	.123	.346	.46	1.29
600	0.18	0.39	.100	.298	.375	1.11
700	0.14	0.305	.072	.208	.269	0.78
900	0.11	0.235	.0533	.1435	.198	0.536
1100	0.10	0.210	.0475	.123	.178	0.460

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°R	$T_o - T_e$					
	2		10		50	
To	W_{Fin}	W_{Total}	W_{Fin}	W_{Total}	W_{Fin}	W_{Total}
552	14.9	256.4	24	152.6	42.2	109.6
571	16.4	230.4	18	114.4	24.4	78
674	1.81	___	7.4	60.8	9.68	33.1
905	.48	42.8	1.82	23.3	1.90	10.4
1068	.128	25.88	.668	12.8	.98	6.58

°R	100		200		300	
To	W_{Fin}	W_{Total}	W_{Fin}	W_{Total}	W_{Fin}	W_{Total}
552	___	___	___	___	___	___
571	57.2	100.7	___	___	___	___
674	11.3	29.7	20.4	37.15	___	___
905	2.22	8.6	3.24	8.25	5.0	9.6
1068	1.02	5.14	1.38	4.59	1.94	4.87

1 kw Per Segment

Fin wt + tube wt + Clad Int   For P_o=.999

^{o}R To	WT FACTOR LB/FT	T_o-T_e 2		10		50	
		W_{Fin}	W_{Total}	W_{Fin}	W_{Total}	W_{Fin}	W_{Total}
552	1.36	14.9	667.9	24	371	42.2	227.2
571	1.28	16.4	606.4	18	306	24.4	168.9
674	0.90	1.81	_____	7.4	139.4	9.68	67.9
905	0.60	0.48	94	1.82	49.2	1.90	20.7
1068	0.54	.128	54.73	.668	26.7	.98	12.9

1 kw Per Segment

Fin wt + tube wt + Clad Int   For P_o=.999

^{o}R To	WT FACTOR LB/FT	100		200		300	
		W_{Fin}	W_{Total}	W_{Fin}	W_{Total}	W_{Fin}	W_{Total}
552	1.36	_____	_____	_____	_____	_____	_____
571	1.28	57.2	175.2	_____	_____	_____	_____
674	0.90	11.3	56.8	20.4	61.9	_____	_____
905	0.60	2.2	16.35	3.24	14.34	5.0	15.2
1068	0.54	1.02	9.82	1.38	8.23	1.94	8.20

$P_o = .999$ 1 kw Per Segment

$^\circ R$	$T_o - T_e$		2		10		50	
T_o	W_{Total}	$\frac{KW}{LB}$	W_{Total}	$\frac{KW}{LB}$	W_{Total}	$\frac{KW}{LB}$		
552	667.9	.003	371	.0054	227.2	.0088		
571	606.4	.0033	306	.0065	168.9	.0118		
674			139.4	.01435	67.9	.0295		
905	94	.0213	49.2	.0406	20.7	.097		
1068	54.73	.0365	26.7	.075	12.9	.155		

$P_o = .999$ 1 kw Per Segment

$^\circ R$	100		200		300	
T_o	W_{Total}	$\frac{KW}{LB}$	W_{Total}	$\frac{KW}{LB}$	W_{Total}	$\frac{KW}{LB}$
552	——	——	——	——	——	——
571	175.2	.0114	——	——	——	——
674	156.8	.035	61.9	.0323	——	——
905	16.35	.122	14.34	.139.	15.2	.132
1068	9.82	.204	8.23	.243	8.20	.244

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$P_o = .99$ 1 kw Per Segment

°R	$T_o - T_e$		2		10		50	
To	W_{Total}	$\frac{KW}{LB}$	W_{Total}	$\frac{KW}{LB}$	W_{Total}	$\frac{KW}{LB}$		
552	256.4	.0078	152.6	.0131	109.6	.0183		
571	230.4	.0087	114.4	.0175	78	.0256		
674	?	?	60.8	.0329	33.1	.0605		
905	42.8	.0466	23.3	.086	10.4	.192		
1068	25.9	.0772	12.8	.156	6.58	.304		

$P_o = .99$ 1 kw Per Segment

°R	100		200		300	
To	W_{Total}	$\frac{KW}{LB}$	W_{Total}	$\frac{KW}{LB}$	W_{Total}	$\frac{KW}{LB}$
552	___	___	___	___	___	___
571	100.7	.01865	___	___	___	___
674	29.7	.0674	37.2	.0538	___	___
905	8.6	.233	8.3	.241	9.6	.208
1068	5.14	.390	4.6	.435	4.87	.410

C. SHELTER ENGINE PERFORMANCE

SOLAR DYNAMIC - FUEL CELL ENGINE SUBSYSTEM EVALUATION

The application considered for this system is a stationary shelter requiring a 5 kw peak and 3.75 kw avg at the load. A total mission of two years is considered with refueling from central storage every 100 hrs. Figure 3-6 is the block diagram of a representative system, incorporating as representative elements the solar stirling engine and a fuel cell system.

Reference parametric curves include:

1. Fig. 3-7 - Typical Polarization Curve.
2. Fig. 3-8 - Specific fuel cell weight versus S.P.C. for typical fuel cells.
3. Fig. 3-9 - Total fuel cell system optimized specific weight versus mission duration.
4. Fig. 3-10 - Interim data on tankage factor versus total useful stoichiometric fuel weight.
5. Fig. 3-11 - Stirling system component weights versus Power output.
6. Table 3-1 - Summary of Storage Battery characteristics
7. Fig. 3-12 - Cryogenic heat sink capacity for subcritical storage
8. Fig. 3-13 - Voltage regulated inverter weight and efficiency
9. Fig. 3-14 - Transformer - Rectifier Converter weight and efficiency
10. Fig. 3-15 - Radiator weights for $\alpha_s = .10$
11. Fig. 3-16 - Radiator weights for $\alpha_s = .30$
12. Fig. 3-17 - Optimum fuel cell system weights
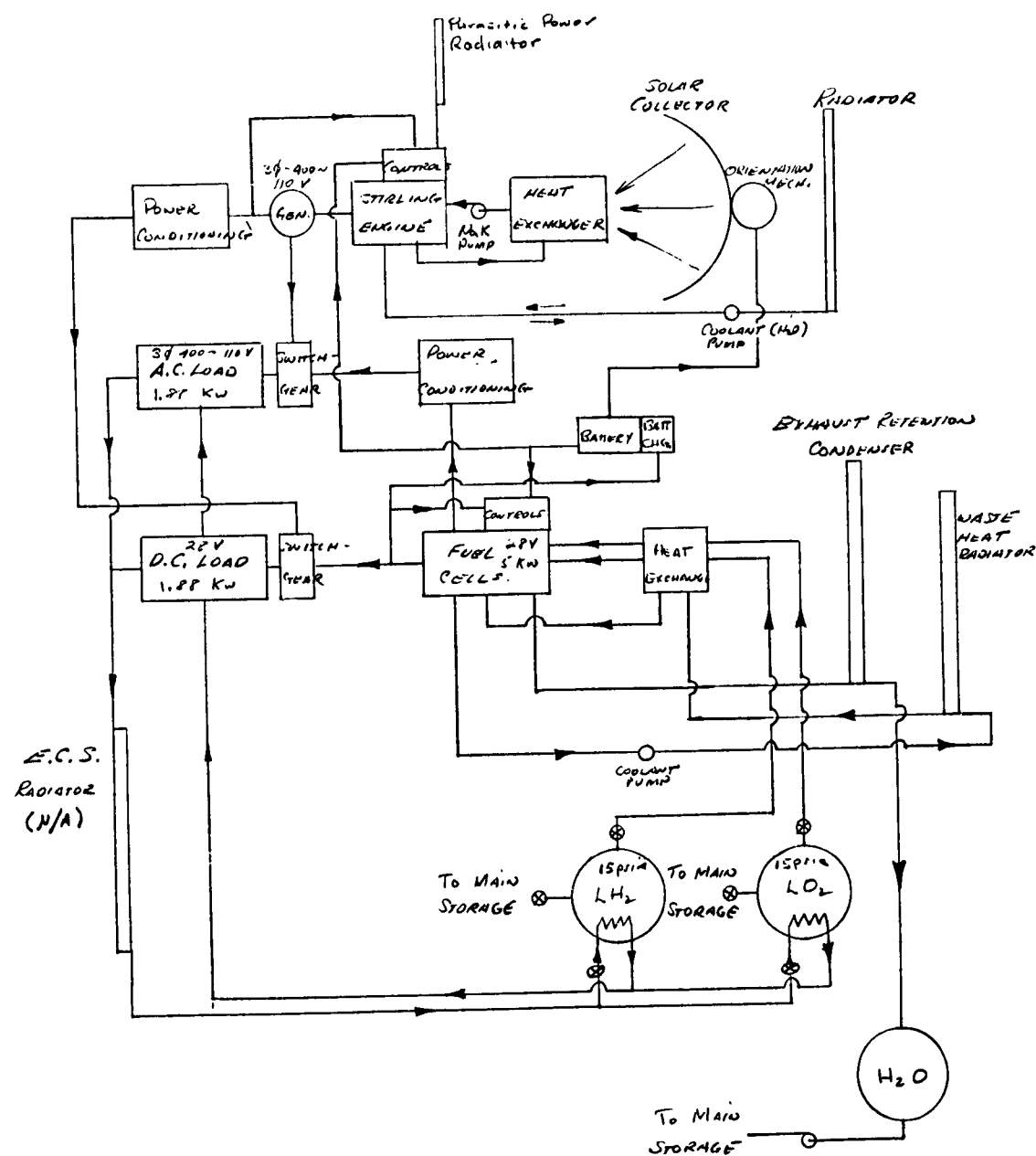
Figure 3-6.   Solar Stirling - Fuel Cell Engine Subsystem
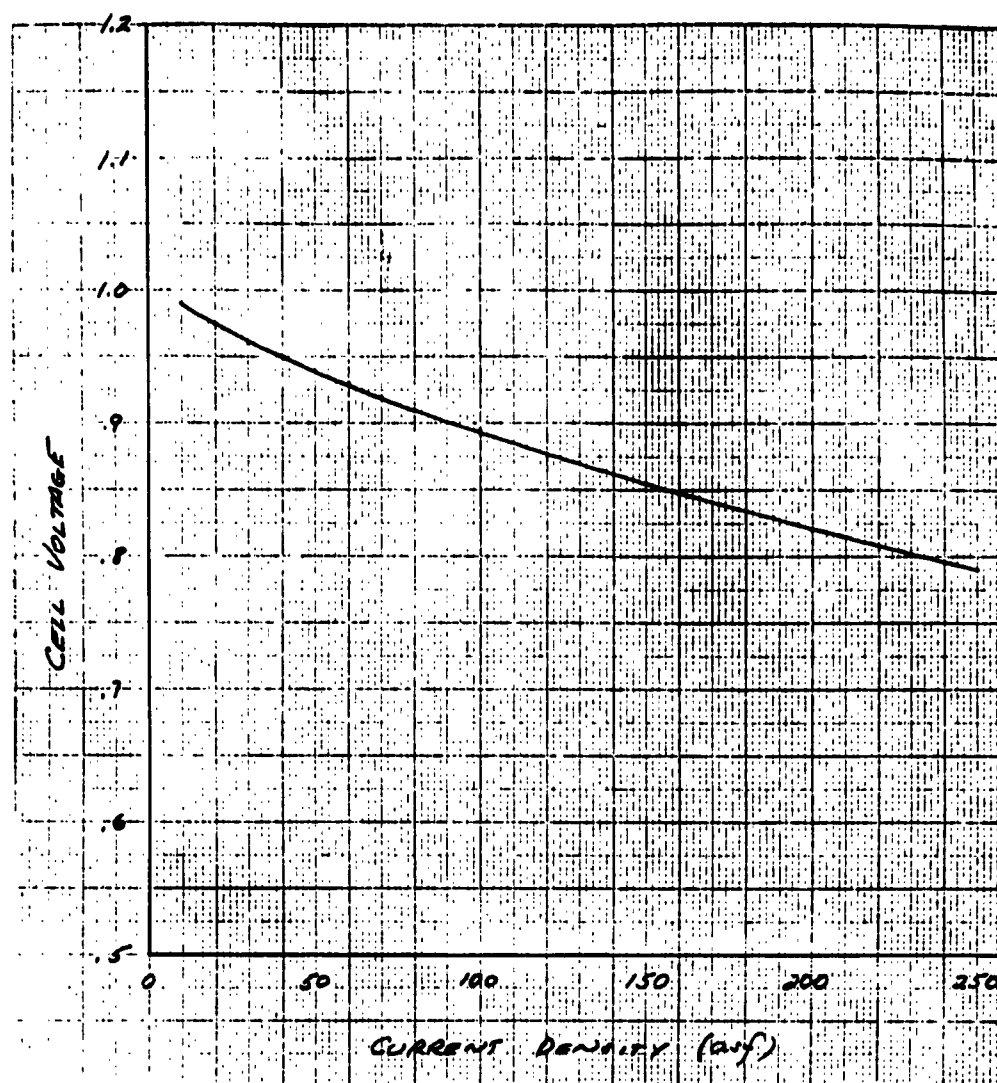
Figure 3-7. Typical Polarization Curve
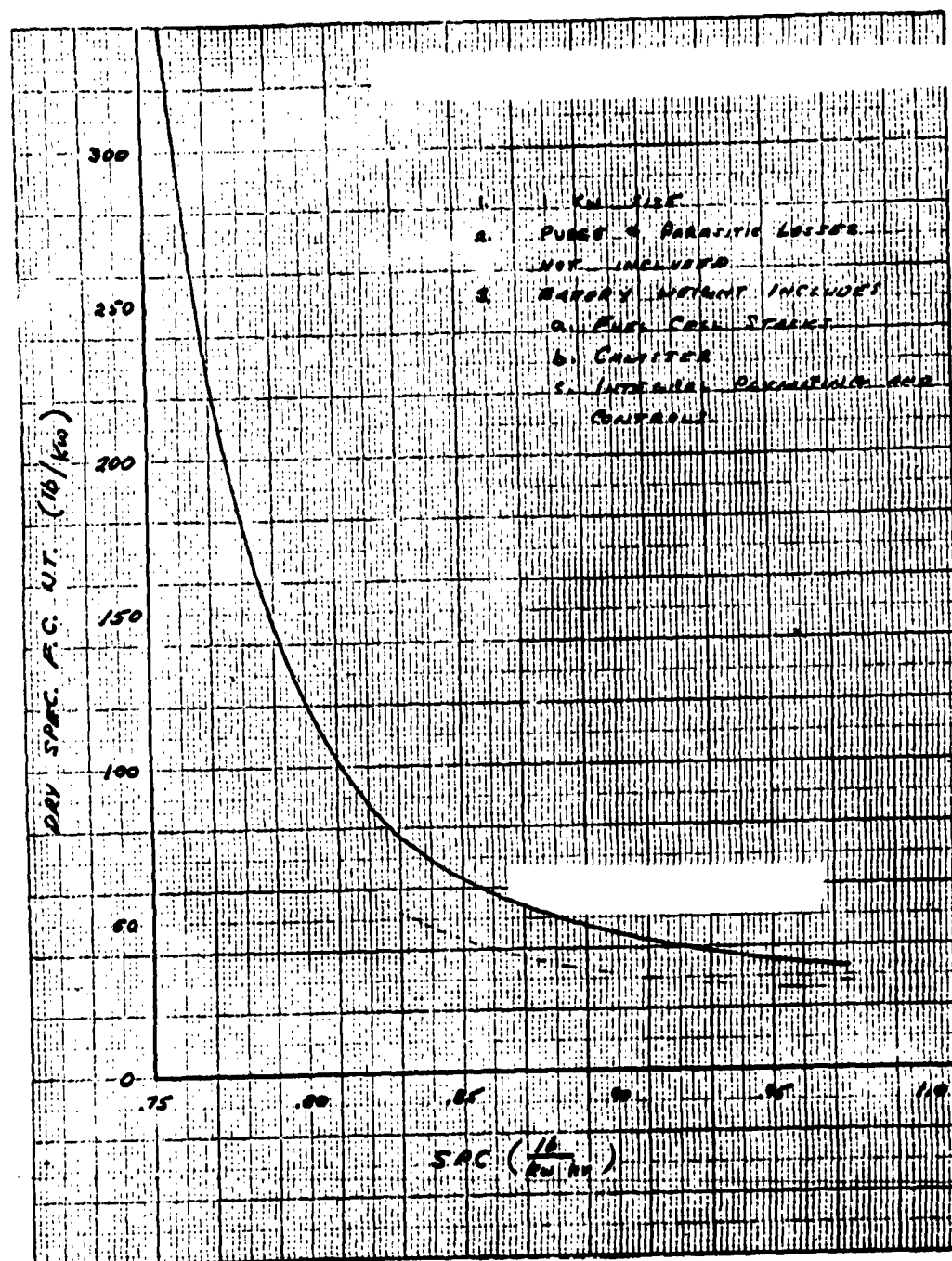
Figure 3-8. Fuel Cell Performance

Figure 3-9

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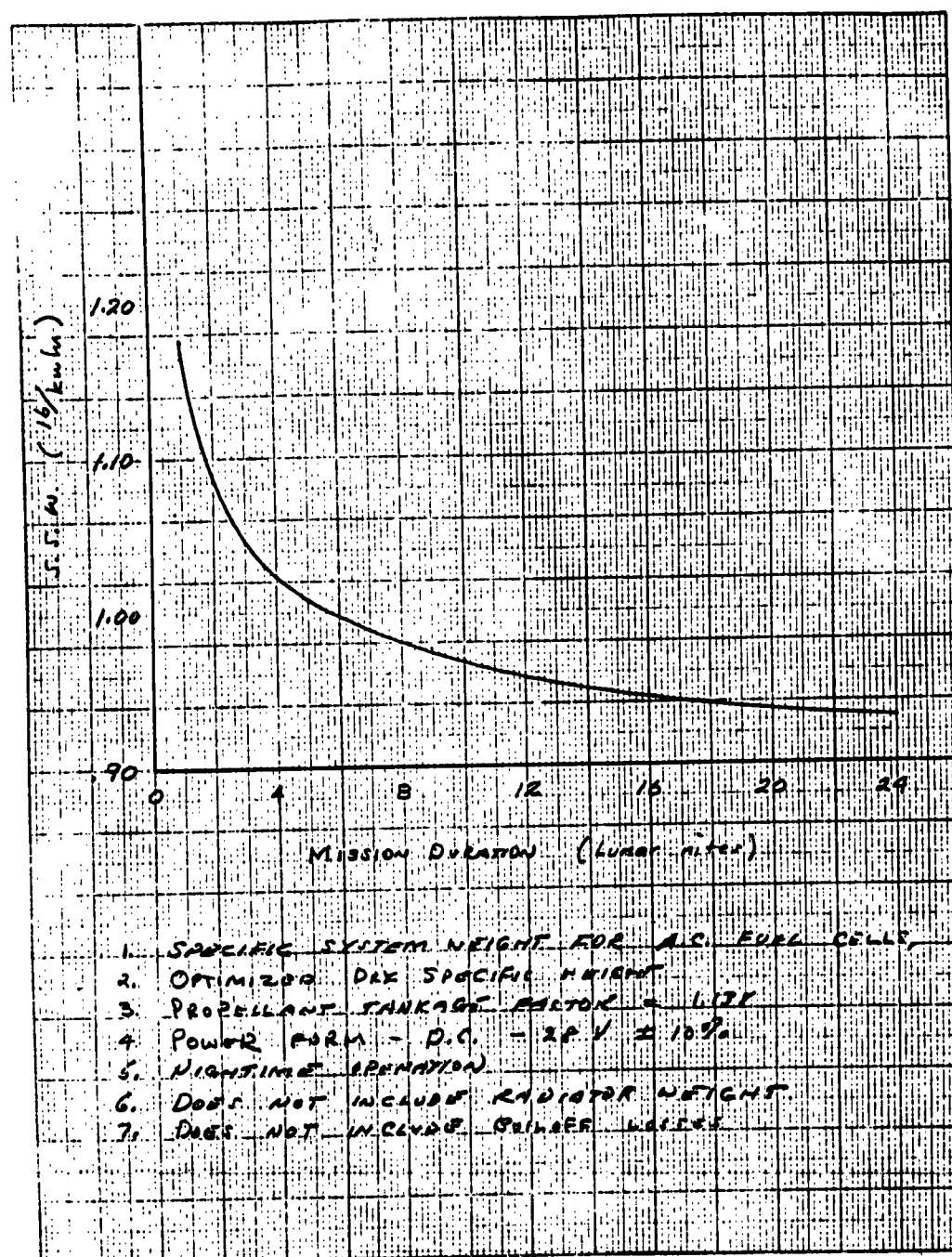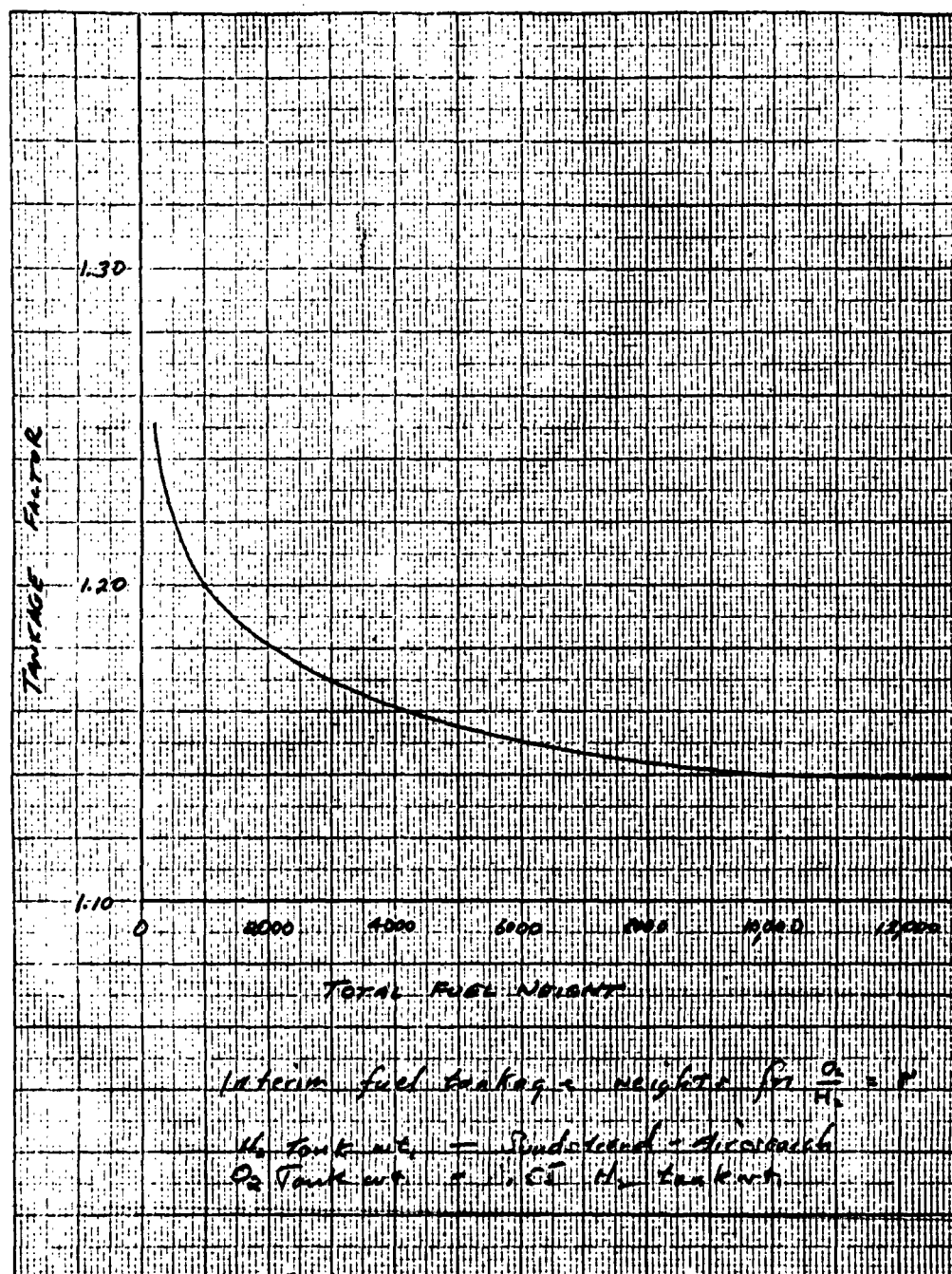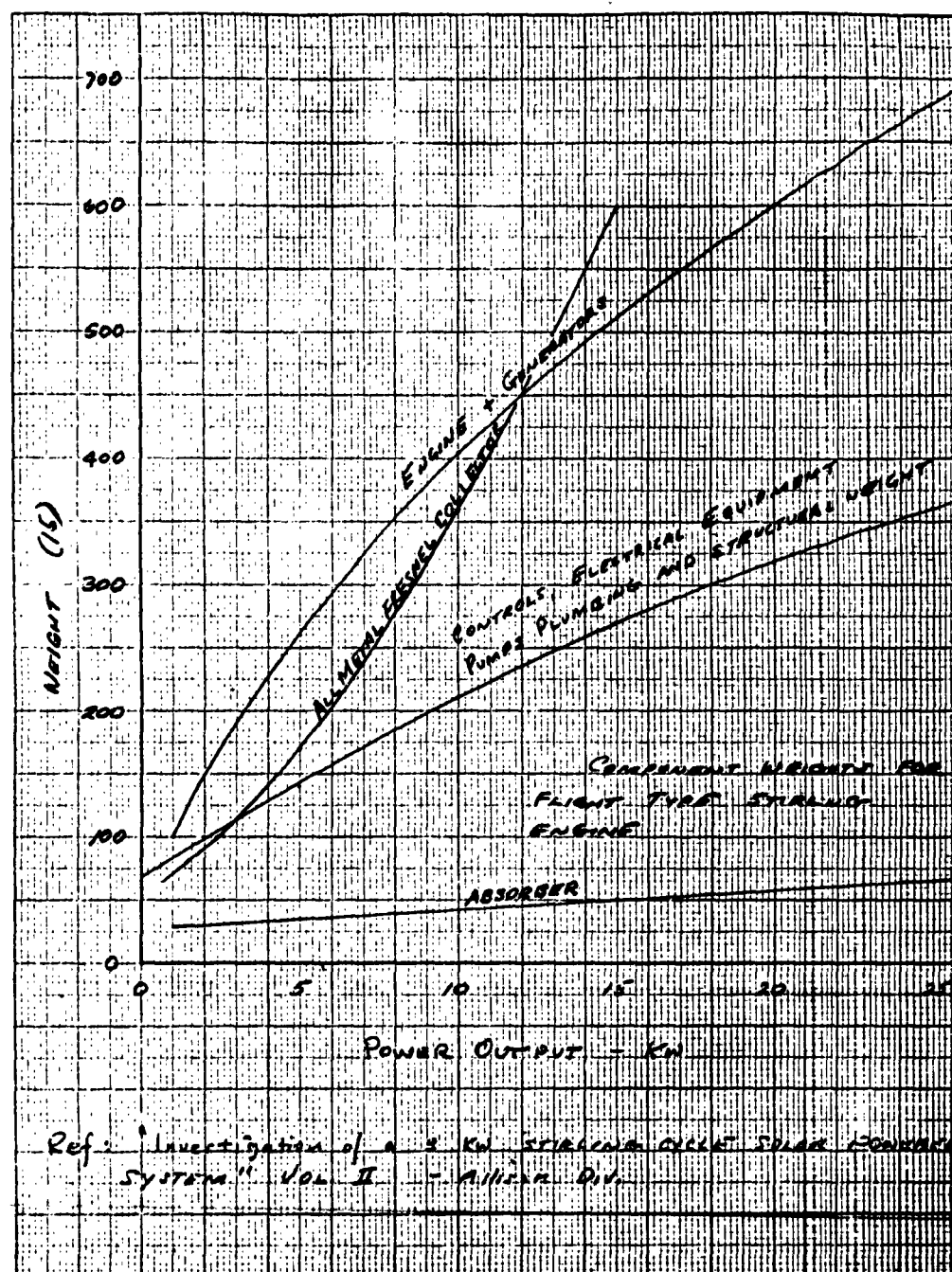
Figure 3-10

Figure 3-11

TABLE 3-1 SUMMARY OF STORAGE BATTERY CHARACTERISTICS

Battery Systems	Units	Lead-Acid $\overline{PbO_2}$ Pb	Nickel-Iron $\overline{Ni(OH)3}$ Fe	Nickel Cadmium $\overline{Ni(OH)3}$ Cd	Silver Zinc $\overline{AgO}$ Zn	Silver Cadmium $\overline{AgO}$ Cd
Characteristics:						
1. Open Circuit Voltage (Low Drain Rate)	Volts	2.05	1.35	1.25	1.86	1.34
2. Theoretical Energy Density (Active Materials)	$\frac{Watt-Hour}{lb.}$	77	119	91	200	124
3. Energy Density (Typical Applications)	$\frac{Watt-Hour}{lb.}$	6-15	9-11	10-12	40-50	25-30
	$\frac{Watt-Hour}{in^3}$	.5-1.3	.6-.7	.6-.9	1.4-2.5	1.3-2.2
4. Approximate Selling Price (5 KW-Hour Capacity)	$/Watt-Hour	.06-.15	.30-.37	.58-.70	.78-1.31	1.04-1..5
5. Recharge Cycles (Typical Applications)	Cycles	10-400	100-300	100-2000	10-300	500-3000
6. Charge Retention	$\frac{\% Loss}{Year}$	20-100	100	20-100	15-20	20-25
7. Shelf Life - Charged (No Maintenance)	Time to 1/2 Charge	Months	Weeks	Months	Years	Weeks

References: Howard, P. L.: "Wet Cell Batteries for Power", Product Engineering: February 15, 1960

Barnartt, S.: "High Energy Batteries", Westinghouse Research Report #6-94781-5-R2

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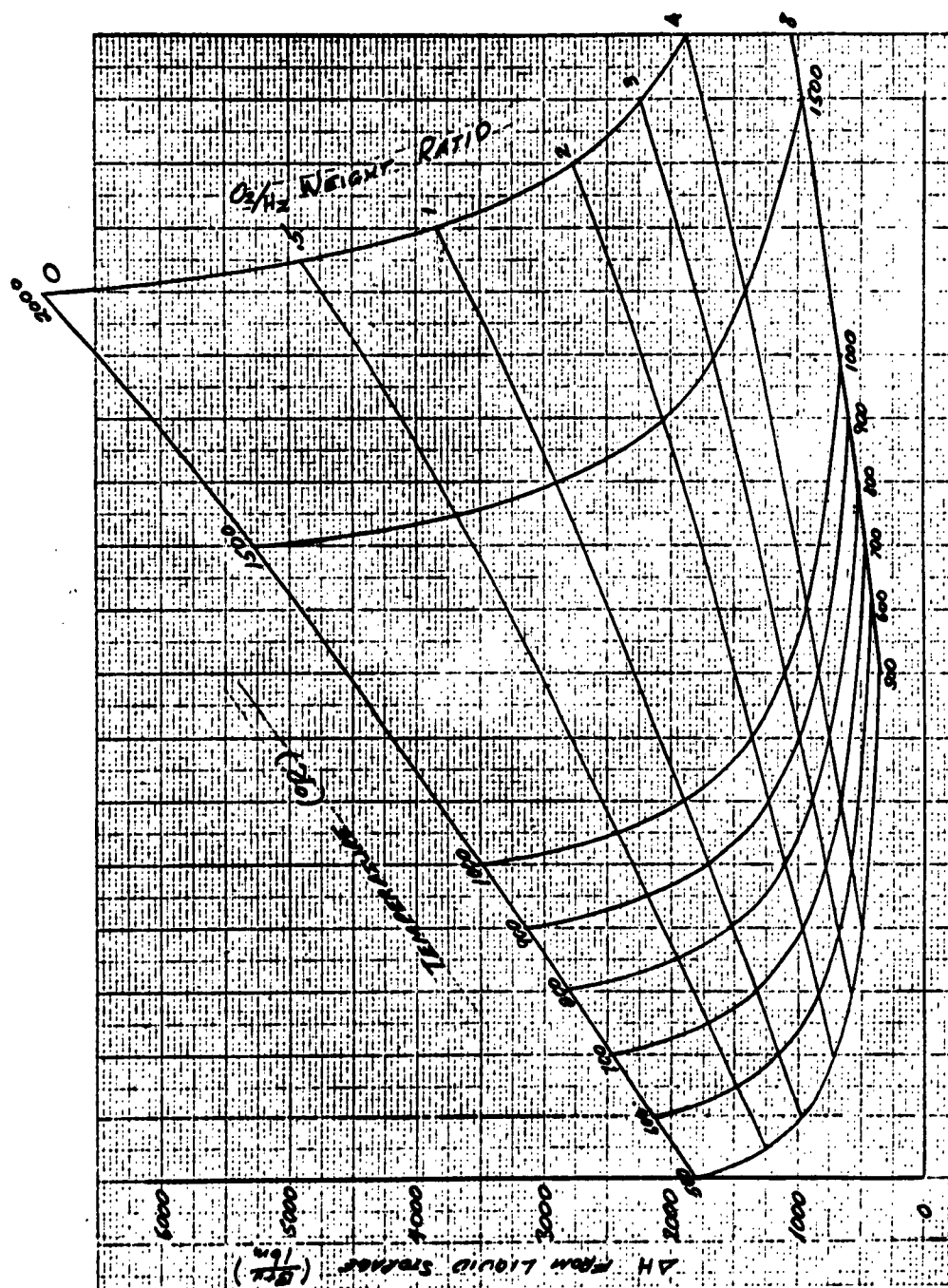
Figure 3-12.   Cryogenic Heat Sink Capacity for
Subcritical Storage of Propellants
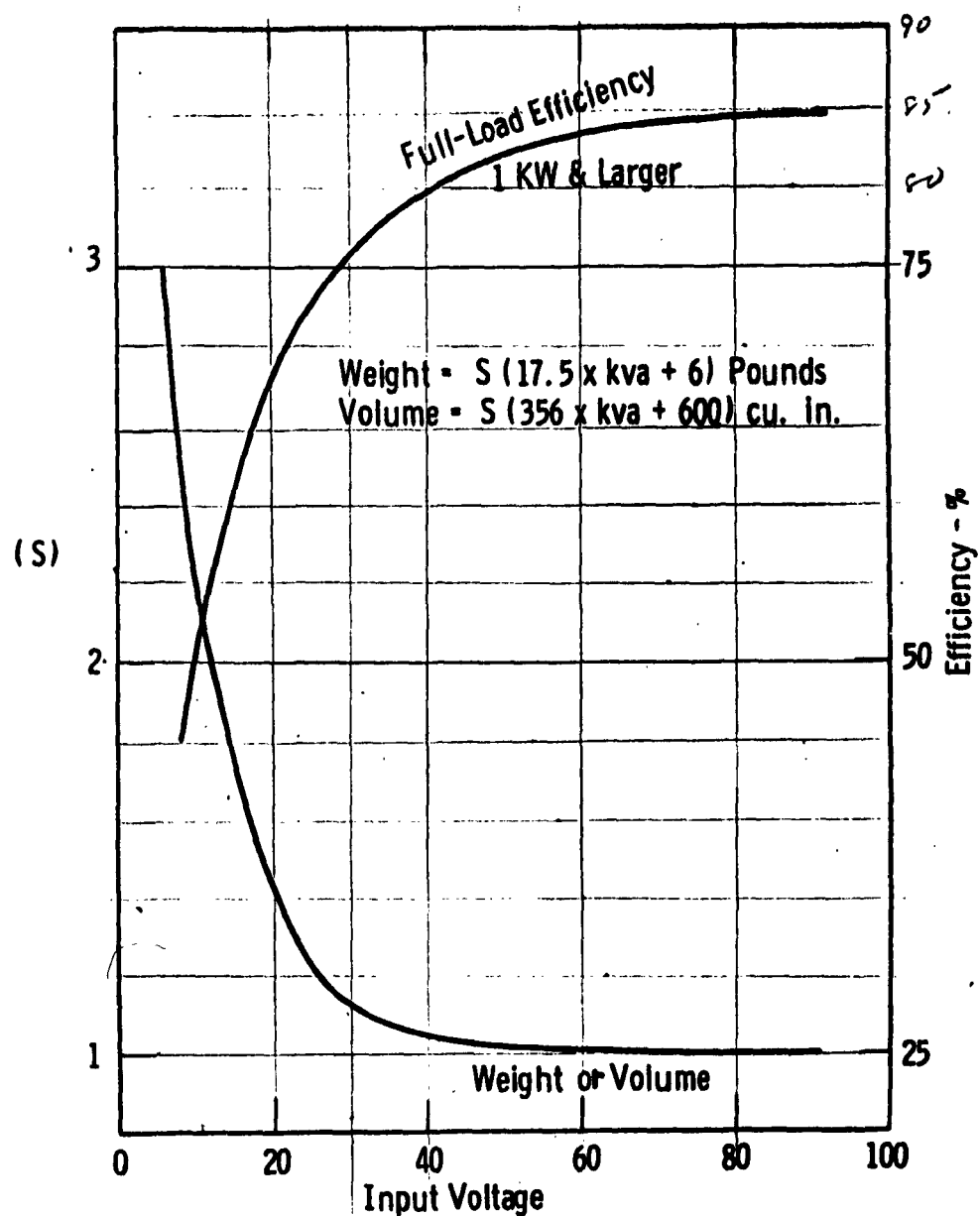
Cold-plate Cooled. Cold plate temperature is 70°F. (Transistor inverters, 3 phase, 400 cps)

Figure 3-13.   Voltage-Regulated Inverter Weight, Volume and Efficiency
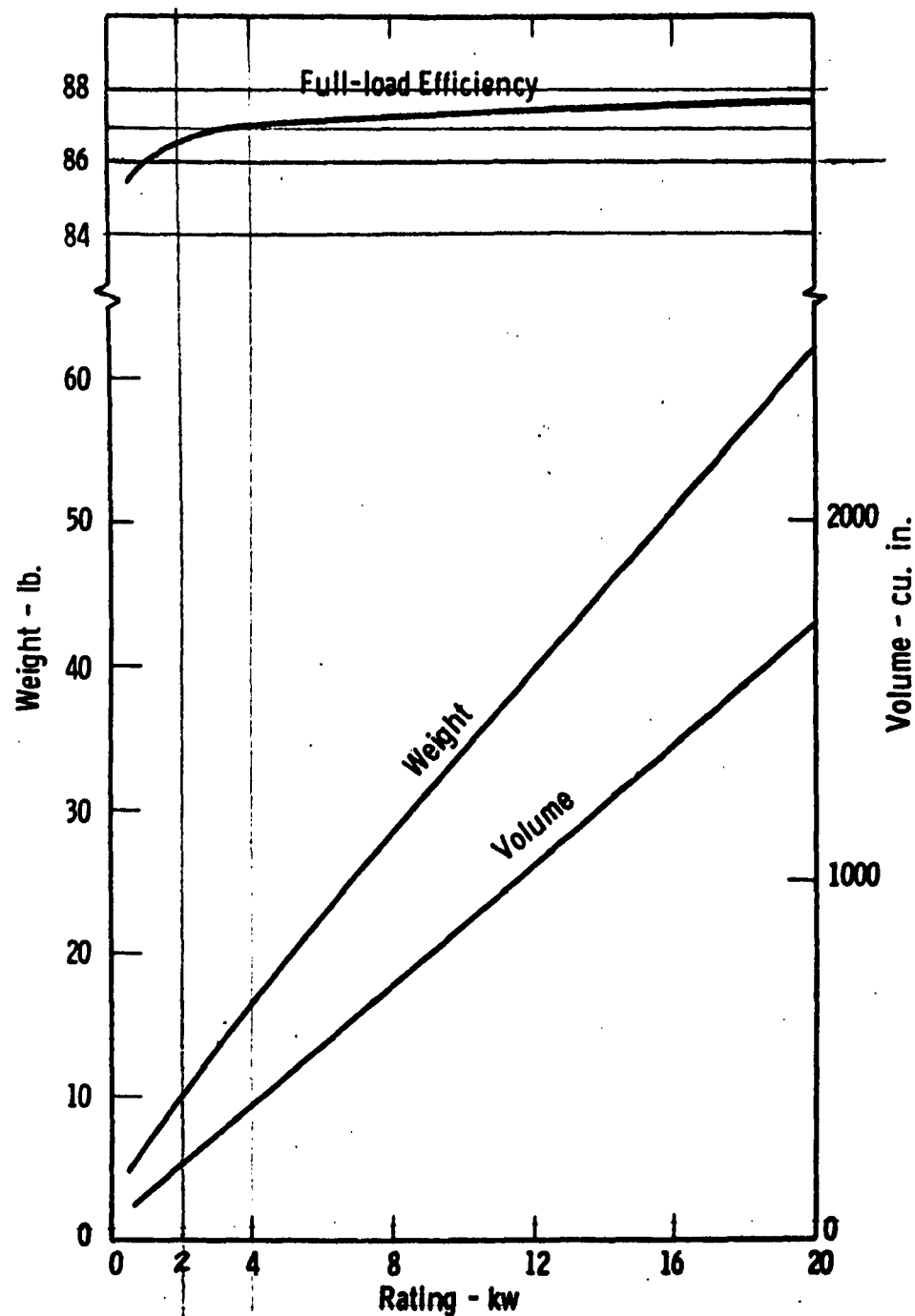
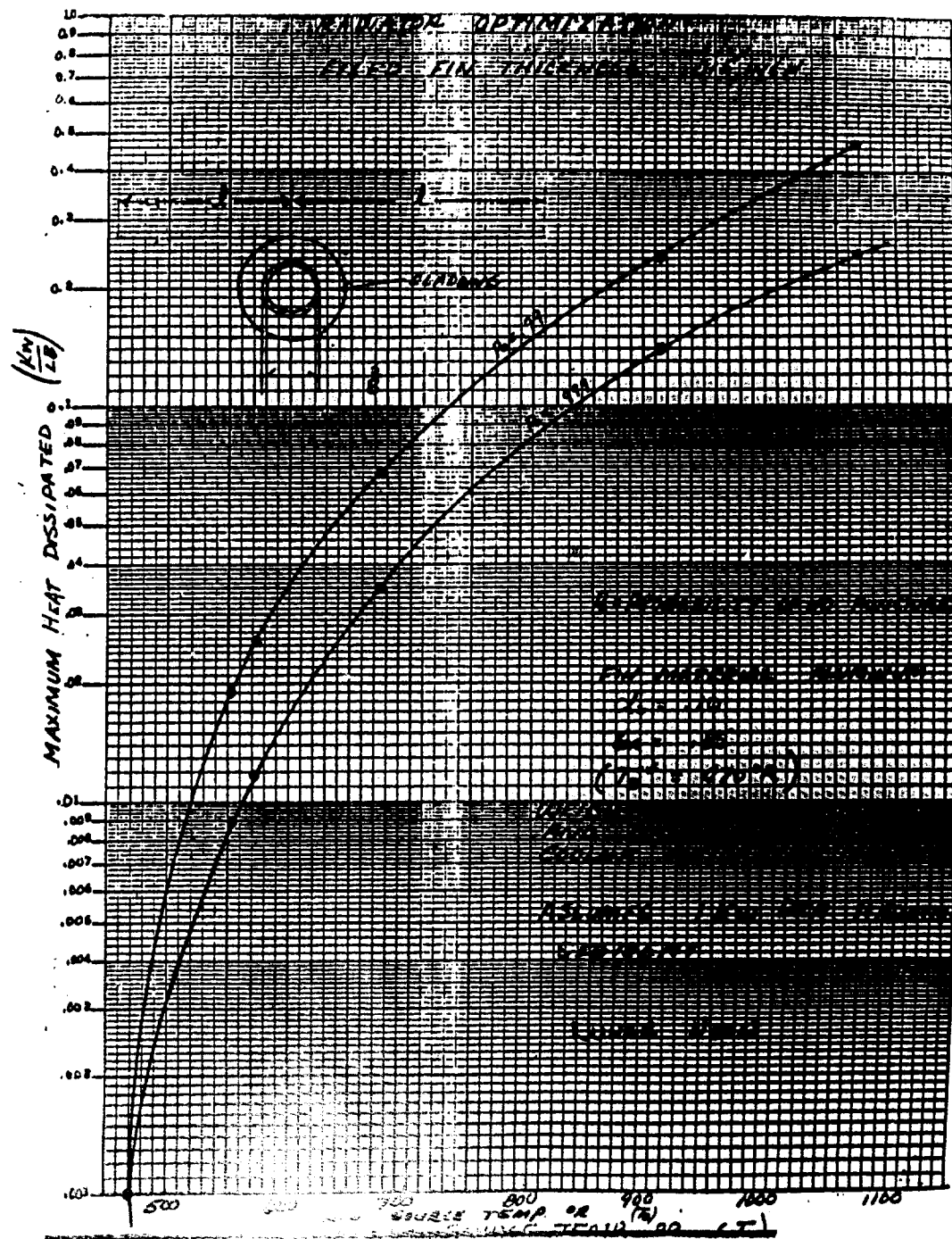Figure 3-14. Transformer-Rectifier Converters

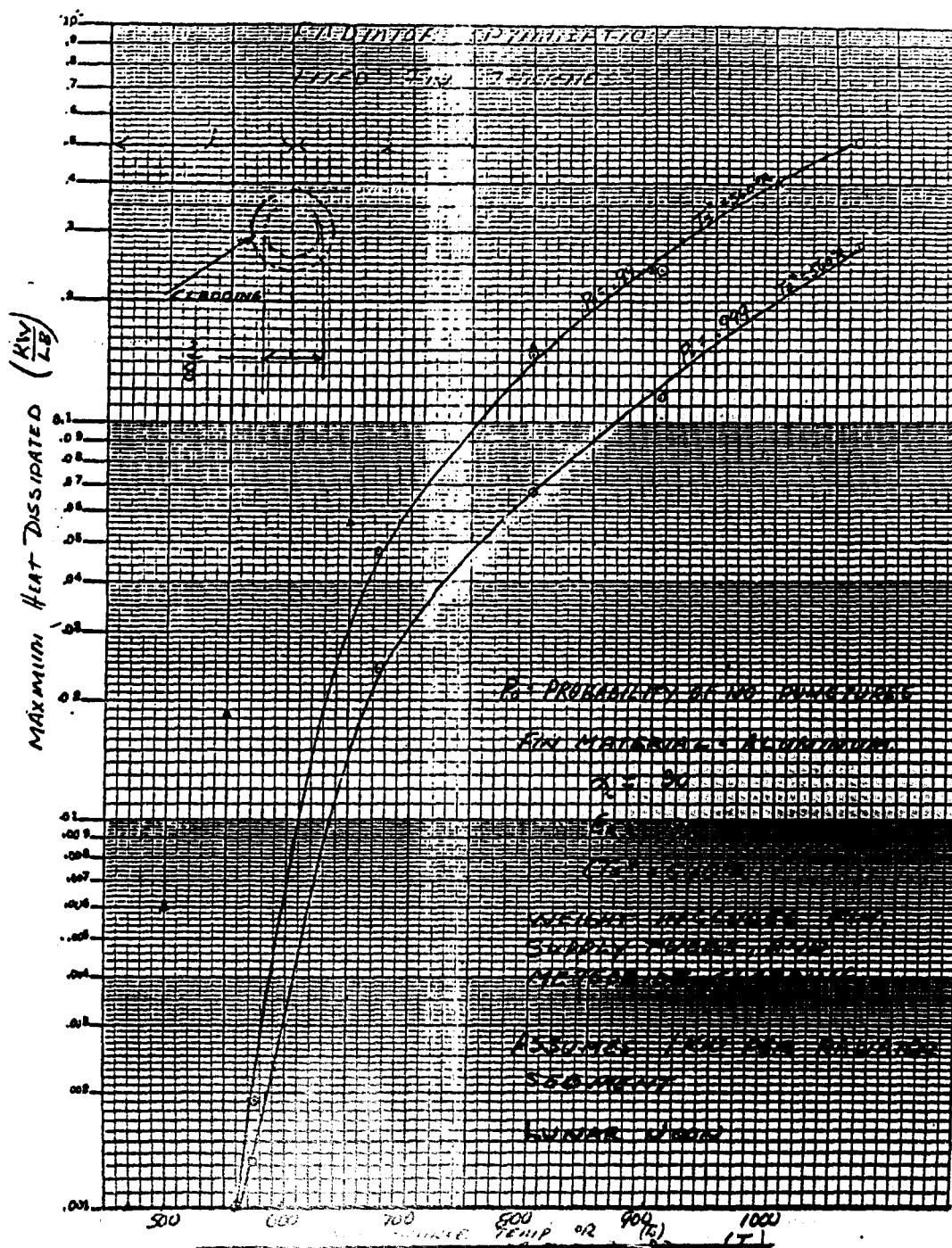Figure 3-72. Radiator Weights for $\alpha_s = 0.10$
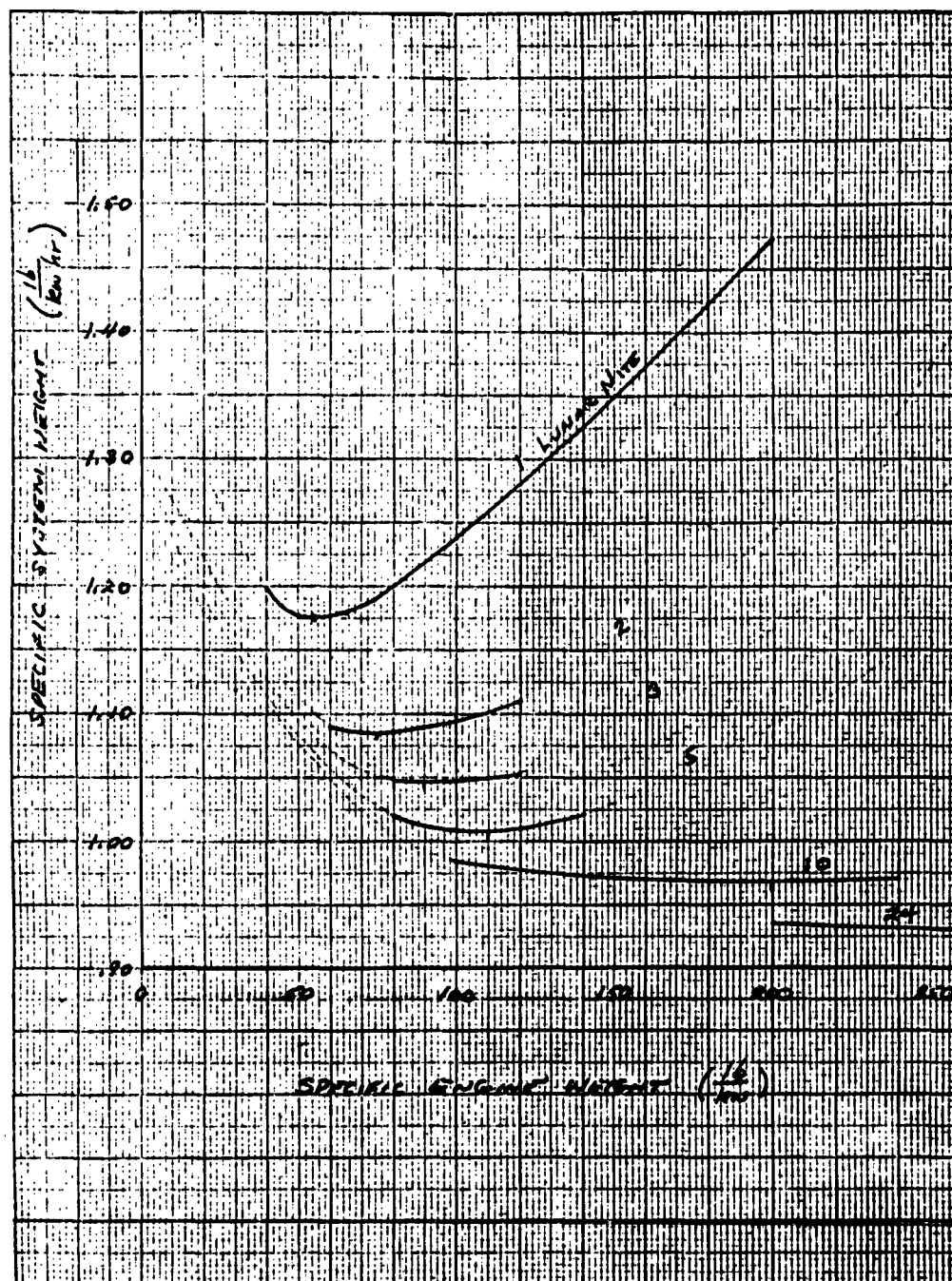
Figure 3-30. Radiator Weights for $\alpha_s = 0.30$

Figure 3-17. Optimum Fuel Cell System Weights

Assumptions

All heat rejection requirements for the solar engine have been satisfied for the conditions existing at the Moon's equator at lunar noon for solar absorptivities of 0.10 and 0.30. The effect on system weight of the load composition (percent dc power required) has been determined for 0.50 and 0.75.

Environmental control system weight has not been included in the total weight, as it has been assumed that the power profile used will include the power required for environmental control.

Fuel cell waste heat rejection has been designed for daytime operation using the optimistic solar absorptivity since for satisfactory crossover at sunrise and sunset the systems might have to run simultaneously.

A purge rate of 1 percent has been assigned to the fuel cell and parasitic power, and is assumed to amount to 2.5 percent of the output.

Determination of the minimum fuel cell system weight was done by finding the minimum weight point of:

(S.P.C.) x (kw . hrs) x (Tankage Factor) + (F.C. Specific Weight) x (kw) = F.C. System Weight.

The results of this calculation are shown in Figures 3-17 and 3-9.

One Synodic day has been taken as 708.7 hrs, with 1.05 hrs required for the sun to rise completely above the horizon. The minimum length of time for the operation of the fuel cells has been taken as 355.4 hrs.

The combined cryogenic propellant tankage factor for subcritical storage was assumed constant at 1.138 for the quantities of propellant being considered.

The solar dynamic engine was designed for the peak kw load required.

An emissivity of 0.85 and a probability of no punctures in three years of 0.999 was used in the radiator design.

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Storage weight penalties for hardware were considered negligible and the loss of propellant during logistic supply has been neglected.

After 14 lunar cycles of operation it has been assumed that all hardware would be replaced.

The system components with fixed weights are:

1. Solar Dynamic System

    Stirling Engine - Generator Assy.

    Solar collector

    Solar absorber

    Stirling Engine controls, electrical, plumbing and structural hardware.

    Waste heat radiator

    Switchgear

    Power conditioner

    Auxiliary battery power pack

2. Fuel Cell System

    On board fuel tanks

    On board product storage

    Waste heat radiator

    Product retention radiator

    Regenerative Heat Exchanger

    Power Conditioner

3. System interconnection weights

    Plumbing

    Electrical Hardware

4. The weights which will vary with mission duration are:

    Fuel Cell Batteries

    Main H_2 Storage Tank + Fuel

    Main O_2 Storage Tank + Fuel

The results of this consideration on a total weight basis for the assumed mission are presented in Figure 3-18. It can be seen that for 75 percent dc power instead of 50 percent dc power, a net weight savings of 2300 lbs (5.6 percent) is achieved. However, for the range of solar absorptivities considered (0.10 and 0.30), the difference is 4100 lbs (9.6 percent) for the same percent dc power.

In the low power range (up to 10 kw) Figure 3-19 is useful in comparing systems with different power requirements. It contains the same information as Figure 3-18 except it has been reduced to specific (per kw) form.

Representative weights of the components of the system are shown in Figure 3-20.

TABLE 3-2

SOLAR STIRLING/A.C. FUEL CELL

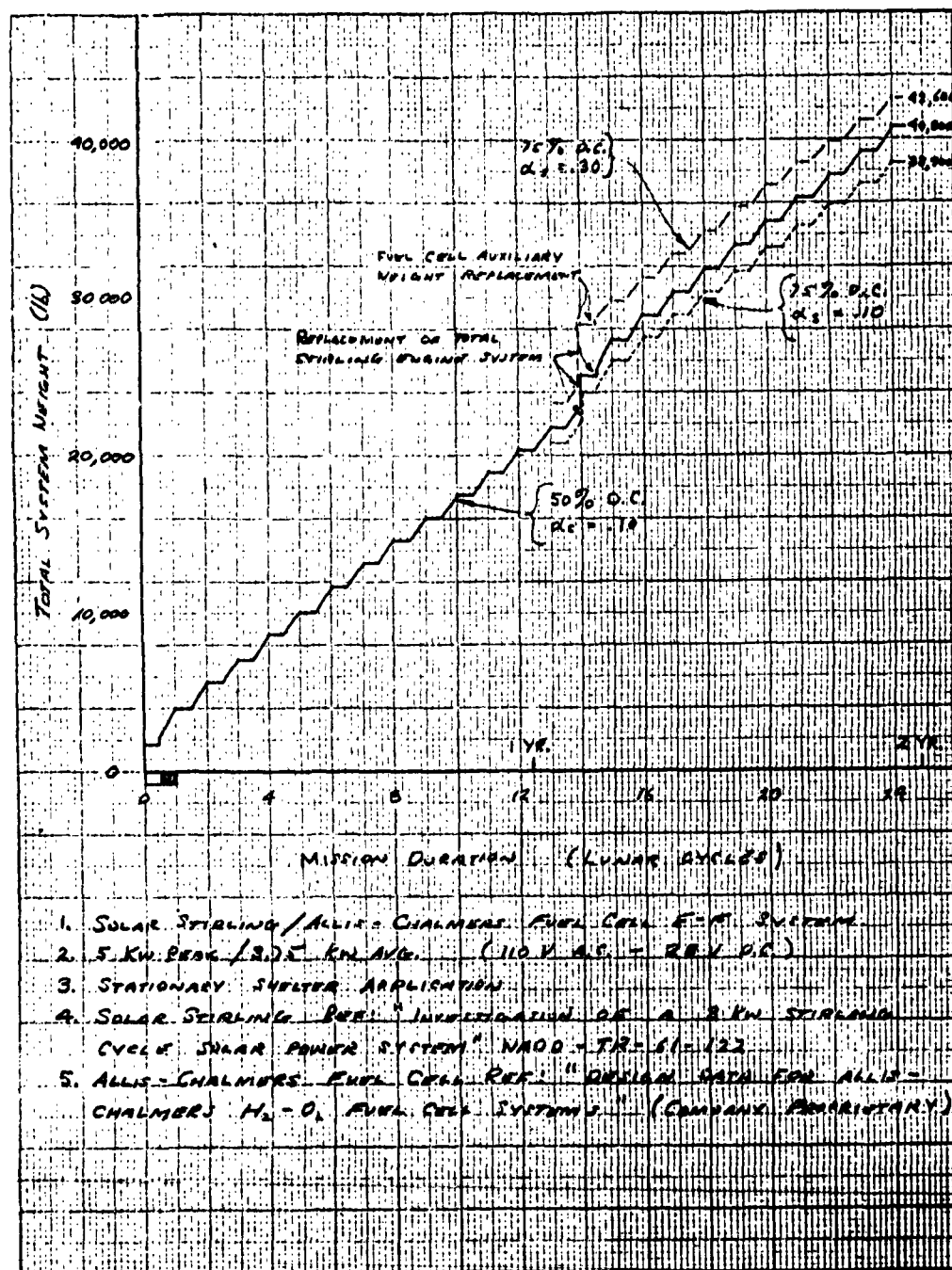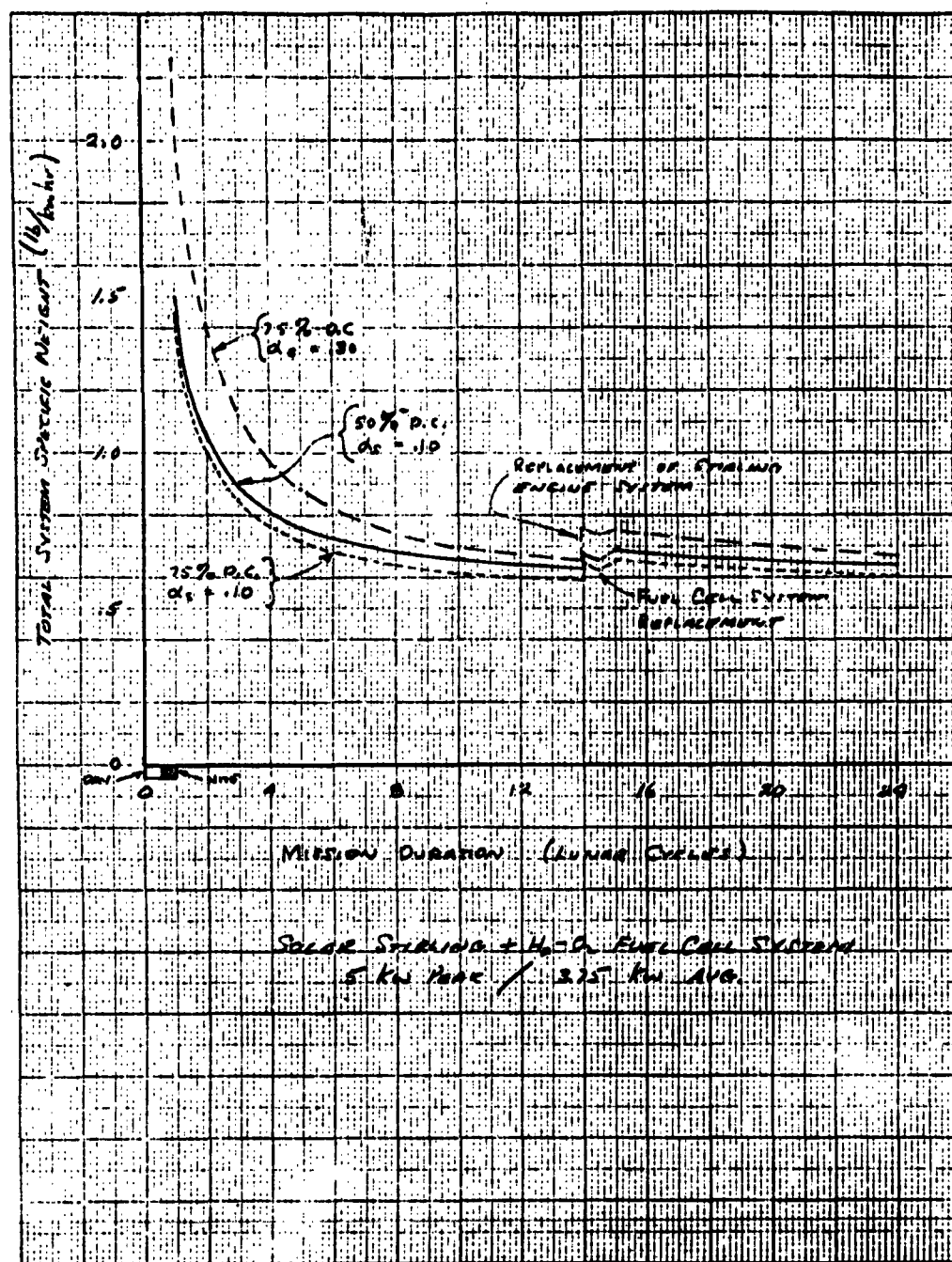|	A	B	C	D
1. Peak Power (kw)	5	5	5	5
2. Avg. Power (kw)	3.75	3.75	3.75	3.75
3. Mission (yrs)	2	2	2	2
4. D.C. Power/2	.50	.75	1.00	.75
5. η A.C. $\longrightarrow$ D.C.	.865	.858	.872	.858
6. η Trans. + Switch	.99	.99	.99	.99
7. η D.C. $\longrightarrow$ A.C.	.75	.75	N/A	.75
8. η Alternator	.80	.80	.80	.80
9. A.C. Load (Avg. kw)	1.88	.94	0	.94
10. D.C. Load (Avg. kw)	1.88	2.81	3.75	2.81
11. A.C. Load (Peak kw)	2.50	1.25	0	1.25
12. D.C. Load (Peak kw)	2.50	3.75	5.00	3.75
13. Stirling A.C. = 11/6	2.53	1.26	0	1.26
14. Stirling D.C. = 12/6 x 5	2.92	4.41	5.79	4.41
15. Peak Stirling Net Power	5.45	5.67	5.79	5.67

Figure 3-18

Figure 3-19. Solar Stirling + H_2 - O_2 Fuel Cell System
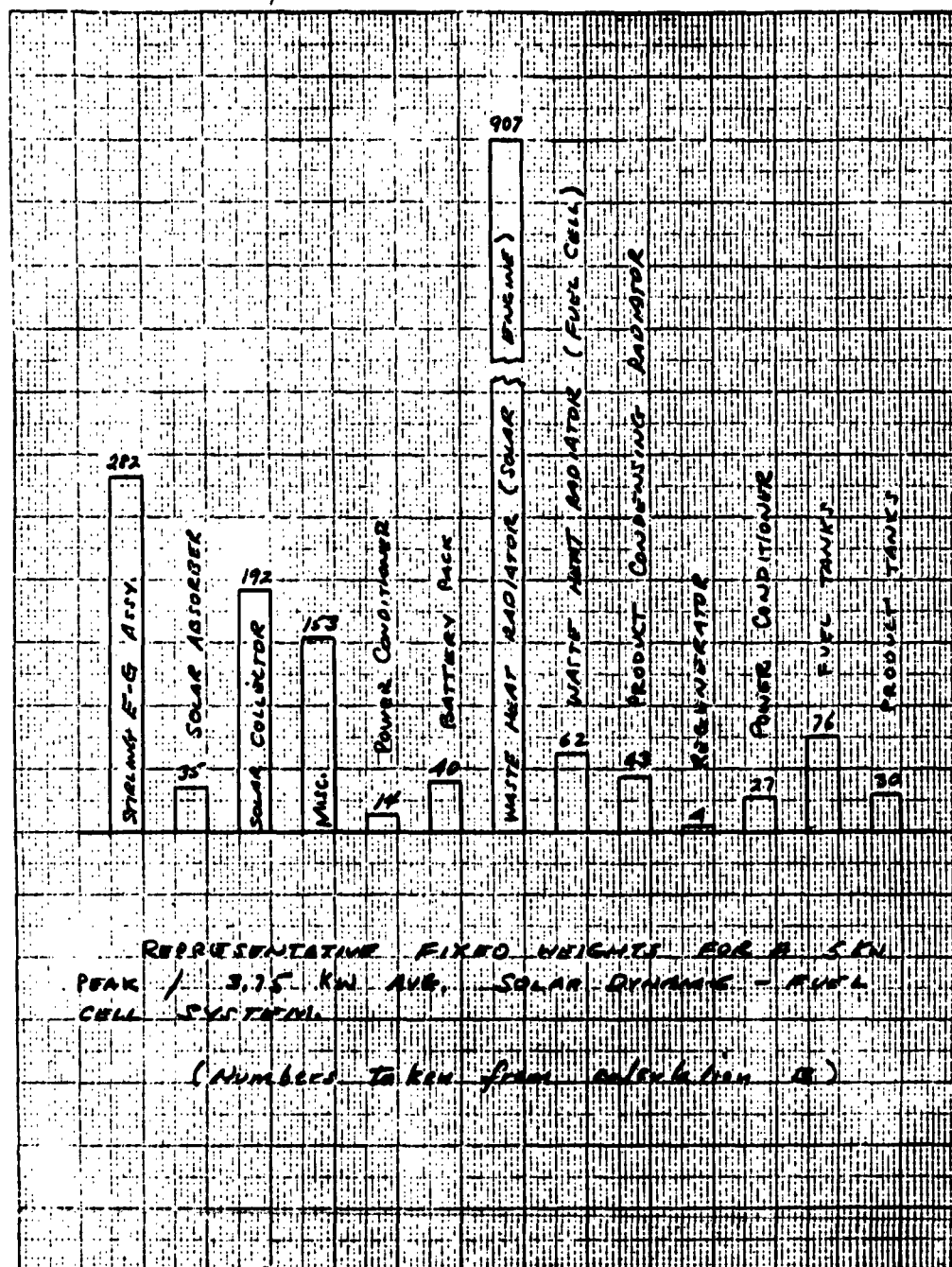
Figure 3-20. Representative Fixed Weights

TABLE 3-2

SOLAR STIRLING/A.C. FUEL CELL (Cont)

	A	B	C	D
16. F.C. Avg. A.C. Load 9/6 x 7	2.53	1.267	---	1.267
17. F.C. Avg. D.C. Load 10/6	1.90	2.84	3.79	2.84
18. Batt. Chg. Load (kw)	.01	.01	.01	.01
19. F.C. Parasitic Load (2.5%)	.05	.07	.09	.07
20. F.C. Gross Avg. Power	4.48	4.19	3.89	4.19
21. Stirling Heat Rej. (kwt)	18.7	19.5	19.9	19.5
22. Stir. Ht. Rej. Temp (^{O}R)	623	623	623	623
23. Solar Absorp. αs	.10	.10	.10	.30
24. Rad. Emissivity ϵ r	.85	.85	.85	.85
25. Po (for 3 yrs.)	.999	.999	.999	.999
26. Spec. Rad. Wt. (lb/Rw)	46.5	46.5	46.5	143
27. Tot. Rad. Wt. 21 x 26	870	907	926	2790
28. F.C. S.P.C.	.918	.800	.800	.800
29. F.C. Heat Rej. W/O Cond. 28 x 5780 x 20/3415 - 20	2.48	1.48	1.37	1.48
30. F.C. Heat Rej. for Cond. 28 x 1057 x 20/3415	1.27	1.037	.988	1.037
31. Temp. of 29 & 30 (^{O}R)	635	635	635	635
32. F.C. Rad. αs	.10	.10	.10	.10
33. F.C. Rad. ϵ s	.85	.85	.85	.85
34. Po (for 3 yrs)	.999	.999	.999	.999
35. Design Point	Noon	Noon	Noon	Noon
36. Sp. Rad. Wt. for 29 + 30	41.6	41.6	41.6	41.6
37. Tot. Rad. Wt. 29 + 36	103	62	57	62
38. Tot. Rad. Wt. 30 + 36	53	43	41	43
39. Stirling E-G Wt.	278	282	288	282
40. Solar Absorber Wt.	35	35	35	35

TABLE 3-2

SOLAR STIRLING/A.C. FUEL CELL (Cont)

	A	B	C	D
41. Solar Collector Wt.	187	192	196	192
42. Stirling Misc. Wt.	150	152	155	152
43. A.C. Switchgear Wt.	1	1	1	1
44. A.C. - D.C. Power Cond. 28 x 20 x 360/3514	10	14	16	14
45. Reg. Capacity (kw)	.42	.35	.33	.35
46. Regen. Wt. 10 x 45	4.2	3.5	3.3	3.5
47. Battery Syst. Wt.	40	40	40	40
48. P.C. Load Sw. Gear (lb)	1	1	1	1
49. D.C. - A.C. Power Cond. "S"	1.15	1.15	1.15	1.15
50. 49 x (17.5 x 9 - +6) (1b)	40	26	0	26
51. Refuel Time (hrs)	100	100	100	100
52. Ideal S.P.C.	.788	.800	.800	.800
53. 52 x 20 x 51 = T.P.C.	353	345	311	345
54. Tankage Factor (54 - 1) 53	1.22	1.22	1.23	1.22
55. Fixed Tank Wts.	77	76	68	76
56. Fixed Prod. Tank Wt. (39→44) + 47 + 27	30	30	30	30
57. Total Stirling Sys. Wt.	1531	1623	1657	3506
58. Total + Misc. Wt. = 57 x 1.15	1760	1870	1900	4040
59. F.C. Syst. Wt. 37 + 38 + 46 + 48 + 50 + 55 + 56	308	242	200	242
60. Tot. F.C. Syst. Wt. 1.15 x 59	354	278	230	278

FOR 1 LUNAR DAY

	A	B	C	D
61. Fuel Cell Syst. Spec. Wt. (lb/Kw-hr)	1.175	1.175	1.175	1.175
62. Fuel Cell Kw-hr . 355 x 20 x 1	1590	1488	1382	1488

TABLE 3-2

SOLAR STIRLING/A.C. FUEL CELL (Cont)

	A	B	C	D
63. Tot F.C. Syst Wt. 61 + 62	1869	1750	1625	1750
64. Fixed Wts. 60 + 58	2114	2098	2130	4268
65. Syst. Tot. Wt. 63 + 64	3983	3848	3755	6018
66. Tot. Kw. Hrs. 709 x 2 x 1	2658	2658	2658	2658
67. Syst. Sp. Wt. #/Kwh	1.497	1.448	1.413	2.26
FOR 2 LUNAR DAYS				
61. Fuel Cell Syst. Spec. Wt. (lb/Kw hr)	1.085	1.085	1.085	1.085
62. Fuel Cell Kw-hrs 355 x 20 x 1	3180	2970	2770	2970
63. Tot F.C. Syst Wt. 61 + 62	3450	3220	3000	3220
64. Fixed Wts. 60 + 58	2114	2098	2130	4268
65. Syst. Tot. Wt. 63 + 64	5564	5318	5130	7488
66. Tot. Kw. Hrs. 709 x 2 x 1	5310	5310	5310	5310
67. Syst. Sp. Wt. #/Kwh	1.048	1.002	.965	1.410
FOR 3 LUNAR DAYS				
61. Fuel Cell Syst. Spec. Wt. (lb/Kw hr)	1.045	1.045	1.045	1.045
62. Fuel Cell Kw-hrs 355 x 20 x 1	4770	4460	4150	4460
63. Tot F.C. Syst Wt. 61 + 62	4980	4660	4330	4660
64. Fixed Wts. 60 + 58	2114	2098	2130	4268
65. Syst. Tot. Wt. 63 + 64	7094	6758	6460	8928
66. Tot. Kw. Hrs. 709 x 2 x 1	7970	7970	7970	7970
67. Syst. Sp. Wt. #/Kwh	.890	.849	.812	1.120

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TABLE 3-2
SOLAR STIRLING/A.C. FUEL CELL (Cont)

	A	B	C	D
FOR 5 LUNAR DAYS				
61. Fuel Cell Syst. Spec. Wt. (lb/Kw hr)	1.008	1.008	1.008	1.008
62. Fuel Cell Kw-hrs 355 x 20 x 1	7950	7430	6910	7430
63. Tot F.C. Syst Wt. 61 + 62	8000	7490	6960	7490
64. Fixed Wts. 60 + 58	2114	2098	2130	4268
65. Syst. Tot. Wt. 63 + 64	10114	9588	9090	11758
66. Tot. Kw. Hrs. 709 x 2 x 1	13300	13300	13300	13300
67. Syst. Sp. Wt. #/Kwh	.760	.721	.684	.884
FOR 10 LUNAR DAYS				
61. Fuel Cell Syst. Spec. Wt. (lb/Kw hr)	.968	.968	.968	.968
62. Fuel Cell Kw-hrs 355 x 20 x 1	15900	14880	13820	14880
63. Tot F.C. Syst Wt. 61 + 62	15400	14250	13260	14250
64. Fixed Wts. 60 + 58	2114	2098	2130	4268
65. Syst. Tot. Wt. 63 + 64	17514	16348	15390	18518
66. Tot. Kw. Hrs. 709 x 2 x 1	26600	26600	26600	26600
67. Syst. Sp. Wt. #/Kwh	.658	.615	.591	.697
FOR 14 LUNAR DAYS				
61. Fuel Cell Syst. Spec. Wt. (lb/Kw hr)	.962	.962	.962	.962
62. Fuel Cell Kw-hrs 355 x 20 x 1	22100	20800	19350	20800
63. Tot F.C. Syst Wt. 61 + 62	21200	20000	18610	20000
64. Fixed Wts. 60 + 58	2114	2098	2130	4268
65. Syst. Tot. Wt. 63 + 64	23314	22098	20740	24268
66. Tot. Kw. Hrs. 709 x 2 x 1	37200	37200	37200	37200
67. Syst. Sp. Wt. #/Kwh	.626	.594	.556	.654

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TABLE 3-2

SOLAR STIRLING/A.C. FUEL CELL (Cont)

	A	B	C	D
At 14 Start Over with 65 as Zero				
68. 65 + 58	25074	23968	22640	28308
69. At 14 1/2 68 + 60	25428	24246	22870	28586
70. At 15 69 + 63 for 1	27297	25996	24495	30058
71. At 16 69 + 63 for 2	28878	27466	25870	31528
72. At 17 69 + 63 for 3	30408	28906	27200	32968
73. At 19 69 + 63 for 5	33428	31736	29780	35798
74. At 24 69 + 63 for 10	40828	38496	36080	42558
75. Kw hrs. @ 68	37200	37200	37200	37200
76. Kw hrs. @ 69	38790	38790	38790	38790
77. Kw hrs. @ 70	39900	39900	39900	39900
78. Kw hrs. @ 71	42500	42500	42500	42500
79. Kw hrs. @ 72	45200	45200	45200	45200
80. Kw hrs. @ 73	50500	50500	50500	50500
81. Kw hrs. @ 74	63800	63800	63800	63800
82. Sp. Syst. Wt. @ 68	.675	.645	.607	.761
83. 14 1/2 69	.655	.624	.590	.737
84. 15 $\sim$ 70	.684	.652	.614	.754
85. 16 $\sim$ 71	.680	.647	.609	.741
86. 17 $\sim$ 72	.672	.640	.602	.730
87. 19 $\sim$ 73	.661	.628	.590	.709
88. 24 $\sim$ 74	.640	.603	.566	.668
89. At 1/2 day 58/1329	1.36	1.41	1.43	1.34

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CALCULATION OF THE SYSTEM WEIGHT OF A SHELTER POWER SUPPLY
USING A STOICHIOMETRIC HYDROGEN AND OXYGEN I.C. ENGINE AND A
SOLAR DYNAMIC ENGINE

Vickers Corporation has proposed (Reference 1) for space applications an internal combustion engine burning a stoichiometric mixture of hydrogen and oxygen. Peak temperatures are limited by condensing the product water vapor and re-injecting it into the cylinder. Such an engine has not yet been built but Vickers has tested a fuel rich hydrogen and oxygen internal combustion engine which at the present point in its development has an SPC of over 2 lbs./kw-hr. Their calculations indicate that a stoichiometric engine could attain a SPC of 1.1 lbs./kw-hr.

Due to the substantial total fuel consumption of any chemical engine over the period of two years (prior to the installation of nuclear power) it is postulated that a shelter would be supplied with power by a solar engine during the lunar day. For the purposes of this analysis a solar stirling engine has been chosen.

Figure 3-21 shows a schematic of a typical system. It has been assumed that the chemical engine would operate for the period of a lunar night plus the time of sunrise (to allow for time to start the solar engine) for a total daily use time of 355.4 hours. Calculations have made for a power level of 3.75 kw average with a SKW peak.

The calculation of radiator weights must necessarily be based on a number of assumptions. The chemical engine, for instance, operates primarily during the night but does operate for a short time early in the lunar day. Its radiator weight, therefore, has been calculated based on radiator results thus far available which are for a solar absorbtivity of 0.10, high noon, and a probability of no punctures in three years of 0.999. The radiator of the solar engine, however, must operate at high noon so its weight has been calculated for values of solar absorbtivity of both 0.10 and 0.30. Radiator

3-86
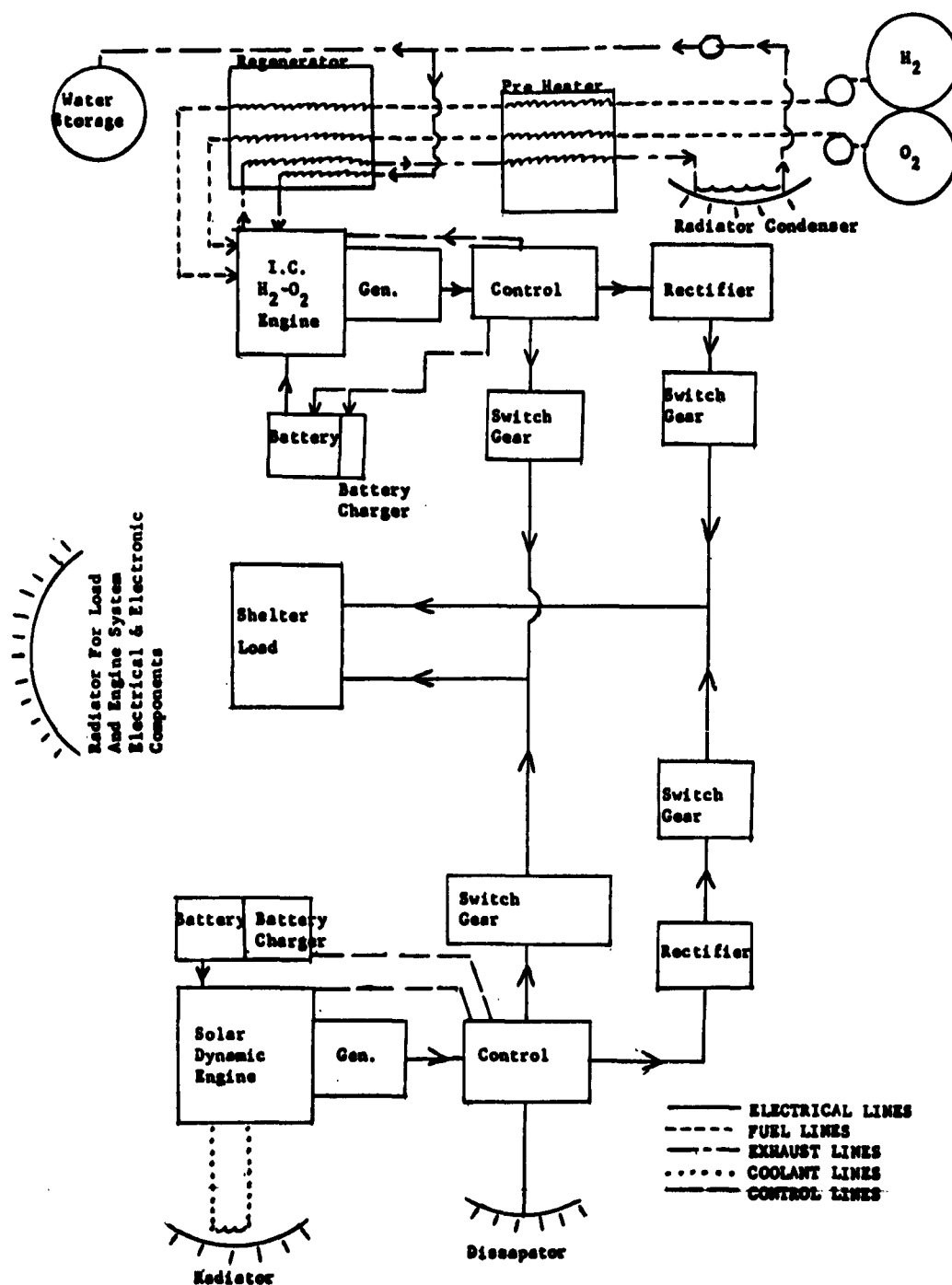
Figure 3-21.   Schematic of Shelter Power Supply

performance was calculated using recently developed information and is shown the section on radiator design.

To find the effect of the type (ac or dc) of power supplied to the load, the performance of a 50-50 dc-ac split and a 25-75 dc-ac split were calculated. The performance and weight of the rectifiers is given in the section on Solar Dynamic-Fuel Cell evaluation.

The propellant tankage factor used was based on Garret data which for stoichiometric hydrogen and oxygen in large quantities is a constant value of 1.138.

Performance is shown in Figures 3-22 through 3-26.

General observations on a shelter power supply using a combination of a stoichiometric H_2-O_2 I.C. engine for night-time power and a solar Stirling engine for day-time power are as follows:

1. The dry weight of the I.C. engine is a relatively small factor for a 2 year mission. Redundancy requirements change this somewhat, but not substantially.

2. Operating efficiency of the chemical engine is paramount. For example, the effect of increasing the SPC from 1.1 /kw-hr to 1.2 /kw-hr is 3,800/lbs. At 5000 $/lb. this amounts to $19,000,000. Even the dc/ac split effects performance substantially. The difference between a 25 percent dc and a 50 percent dc supply is approximately 1600 lbs. for a 2 year mission. This emphasizes the importance of ultra high efficiency conversion equipment for use on the moon.

3. For low temperature rejection equipment the value of the solar absorptivity of the radiator is critical. For instance, if the value is as high as 0.3, the weight of the radiator of the stirling solar engine is approximately 2800 lbs., which is about four times the weight of the engine.
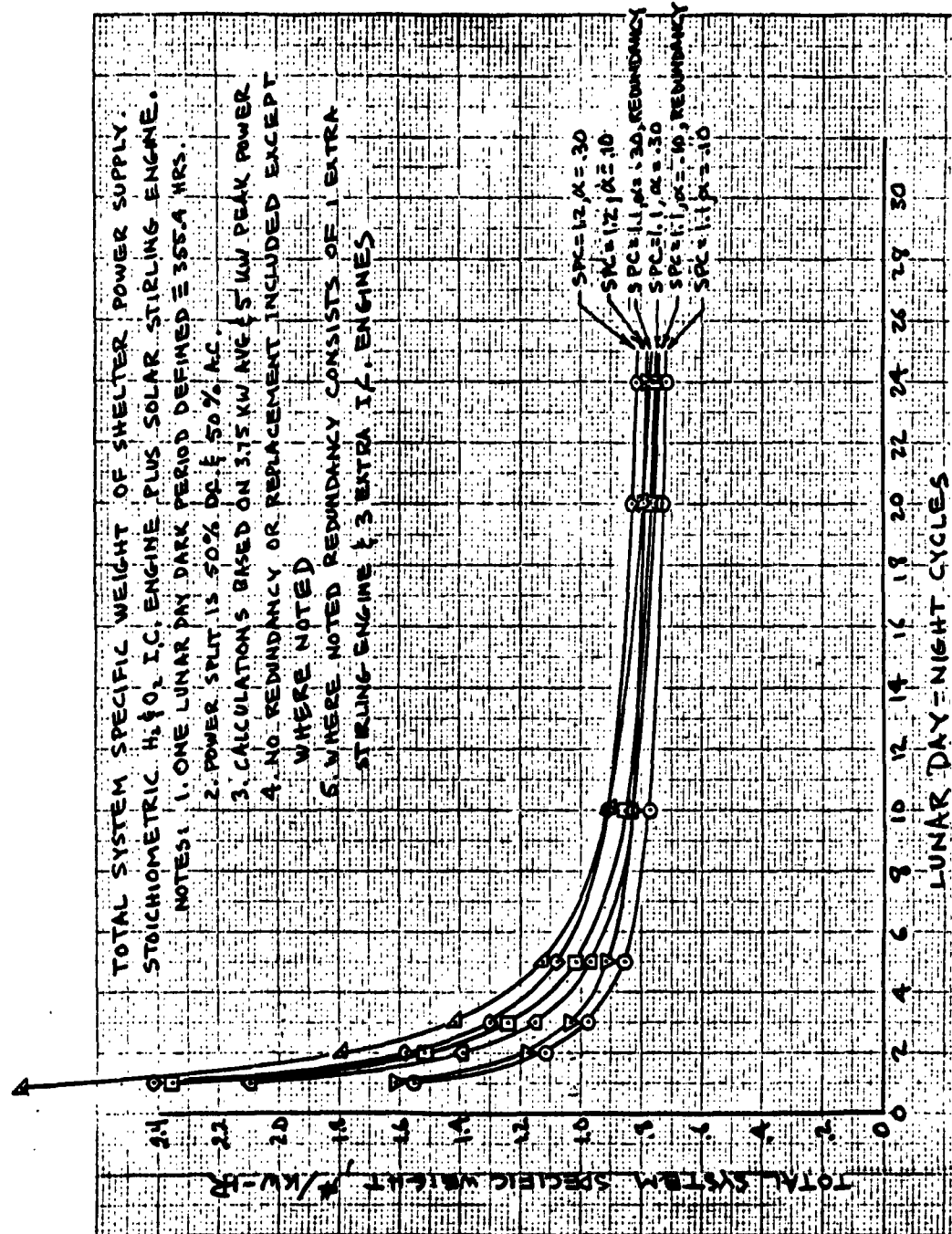
Figure 3-22. Total System Specific Weight of Shelter Power Supply

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TOTAL SYSTEM WEIGHT OF SHELTER POWER SUPPLY FOR 3.15KW AVG, 5KW PEAK. STOICHIOMETRIC H₂ & O₂ I.C. ENGINE PLUS SOLAR STIRLING ENGINE

NOTES: 1. ONE LUNAR DAY DARK PERIOD DEFINED = 355.4 HRS
2. POWER SPLIT IS 50% DC & 50% AC
3. NO REDUNDANCY OR REPLACEMENT INCLUDED EXCEPT WHERE NOTED.
4. WHERE NOTED REDUNDANCY CONSISTS OF 1 EXTRA STIRLING ENGINE & 5 EXTRA I.C. ENGINES.

51,848#, SPC=1.2, α=.3
47,720# SPC=1.2, α=.3
49,949# SPC=1.1, α=.3, REDUNDANCY
47,949# SPC=1.1, α=.3
47,261# SPC=1.1, α=.3, REDUNDANCY
45,820, SPC=1.1, α=.1

FOR ABOVE SYSTEMS IN ORDER SHOWN:

	DRY WEIGHT	2 YEAR WEIGHT REACTANTS PLUS TANKAGE	TOTAL
	4,448	47,400	51,848
	2,320	47,400	49,720
	5,889	43,500	49,389
	4,449	43,500	47,949
	3,761	43,500	47,261
	2,320	43,500	45,820

LUNAR DAY-NIGHT CYCLES

TOTAL SYSTEM WEIGHT, POUNDS

Figure 3-23. Total System Specific Weight of Shelter Power Supply

3-90
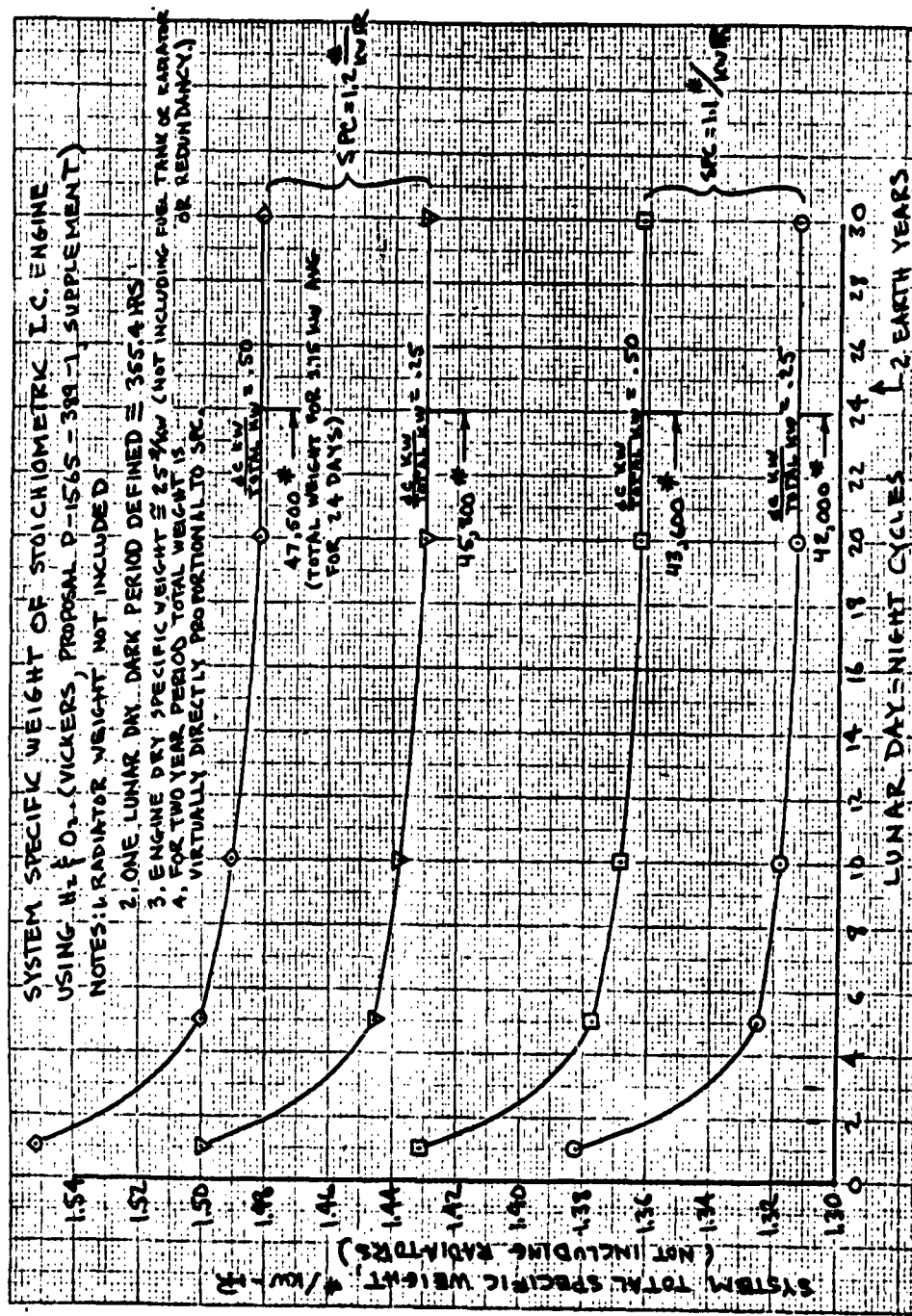
Figure 3-24. System Specific Weight of Stoichiometeric I.C. Engine

Figure 3-25. Shelter Power Supply Component Weights
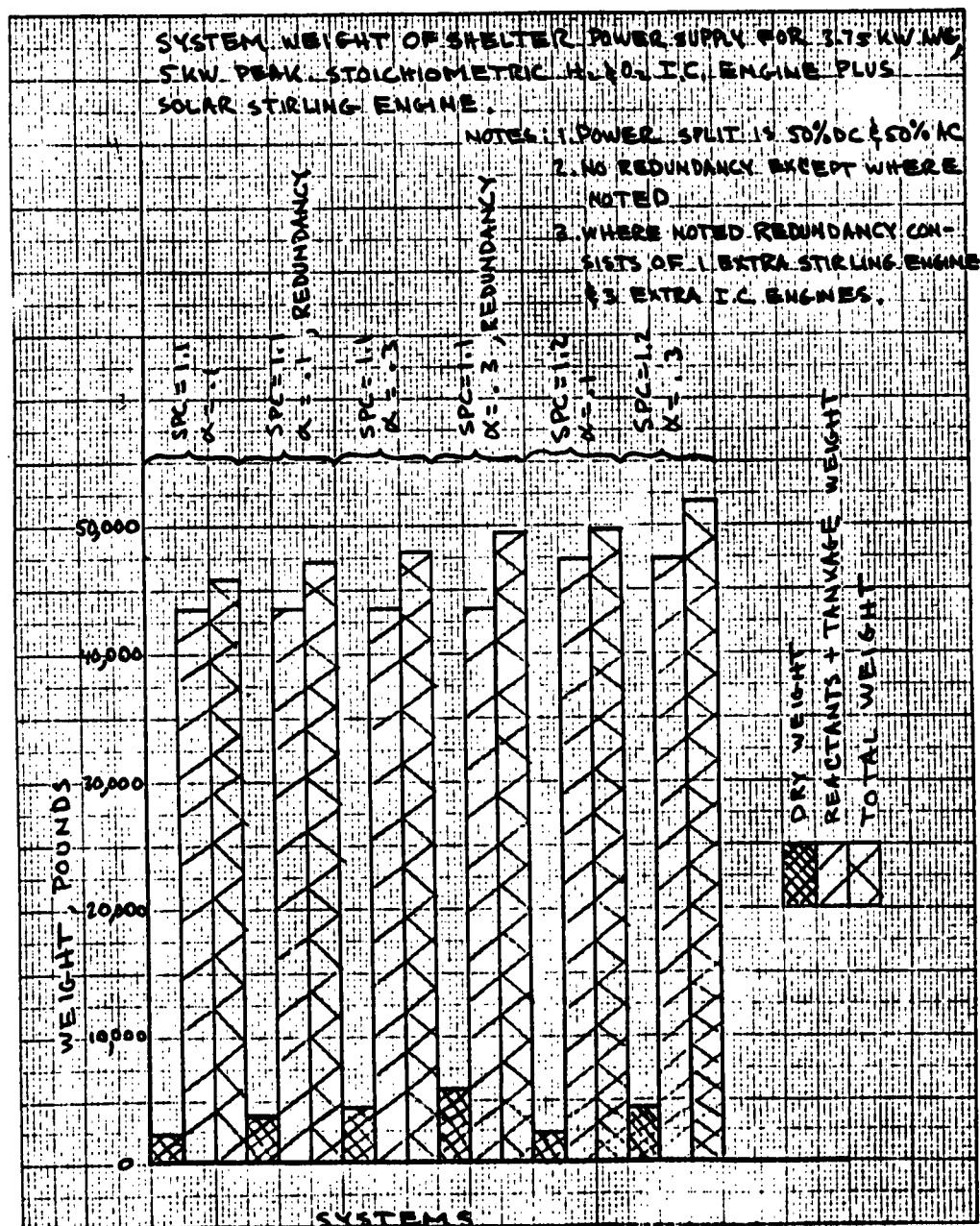
Figure 3-26. System Weight of Shelter Power Supply

REFERENCES

N. E. Morgan. Stoichiometric Hydrogen-Oxygen Space Power System. Proposal P1565-389-1. Vickers Corporation. (Company Proprietary)

WADD-TR-61-122, VOLUME II. Investigation of a 3IW Stirling Cycle Solar Power System. Allison Division of General Motors.

Design Data for Allis Chalmers H_2-O_2 Fuel Cell System. (Company Proprietary).

TABLE 3-3

PERFORMANCE CALCULATION OF CHEMICAL DYNAMIC
ENGINE FOR SHELTER APPLICATION
STOICHIOMETRIC I.C. ENGINE

1. Avg. Power, kw	3.75	3.75	3.75	
2. Peak Power, kw	5.0	5.0	5.0	
3. Avg./Peak = 1/2	.75	.75	.75	
4. DC Power/Total Power (Assumed) (Fig 23)	.25	.50	.25	.50
5. η Rectifier @ 1 x 4 (Lima Data)	.86	.865	.86	.865
6. η Switchgear & Transmission (Assumed)	.99	.99	.99	.99
7. AC Power for Avg. dc Power = 1 x 4/5 x 6 kw	1.1	2.19	1.1	2.19
8. Battery Charging Power, kw (Estimated)	.01	.01	.01	.01
9. Total Avg. ac Power = 7 + $\left[(1\text{-}4) \times 1\right]$ + 8, kw	3.925	4.075	3.925	4.075
10. η Rectifier @ 100% Power (@ 2 x 4)	.865	.865	.865	.865
11. AC Power for Peak DC Power $\cong$ 2 x 4/10, kw	1.443	2.89	1.433	2.89
12. Total Peak ac Power $\cong$ 11 + $\left[(1\text{-}4) \times 2\right]$ + 8, kw	5.203	5.40	5.203	5.40
13. Avg. ac Power/Peak ac Power $\cong$ @/12	.754	.755	.754	.755
14. BSPC @ 9, #/kw-hr (Projected)				
15. η generator (Projected)				

TABLE 3-3 (Cont)

16. Net SPC @ 9 = 14/15, #/kw-hr	1.1	1.1	1.2	1.2
17. Lunar Day + Sunrise, hrs	355.4	355.4	355.4	355.4
18. Fuel Consump. per lunar day = 9 x 16 x 17, #	1535	1592	1673	1735
19. Fuel Consump. per lunar day /kw = 18/1	409	425	446.5	462.5
20. Tankage Factor = $\dfrac{\text{wet system wt}}{\text{Fuel wt}}$, (Garret Data)	1.138	1.138	1.138	1.138
21. Fuel System wet wt = 20 x 18, #/day	1748	1812	1907	1975
22. Fuel System wet wt = 21 x 2, #	3550	3624	3814	3950
23. Fuel System wet wt = 21 x 3	5245	5440	5720	5930
24. Fuel System wet wt = 21 x 5	8740	9060	9540	9880
25. Fuel System wet wt = 21 x 10	17480	18120	19070	19750
26. Fuel System wet wt = 21 x 20	35500	36240	38140	39500
27. Fuel System wet wt = 21 x 24	41950	43500	45800	47400
28.				
29.				
30.				
31.				
32.				

COMPONENT DRY WEIGHTS FOR CHEMICAL DYNAMIC ENGINE (STOICHIOMETRIC I.C.) PLUS SOLAR STIRLING FOR SHELTER APPLICATION

33. I.C. Engine Sp. Wt., #/kw (Vickers information)	18	18	18	
34. I.C. Engine Total Wt = 33 x 12, #	93.7	97.3	93.7	97.3
35. Rectifier wt @ 4 x 2, # (W data)	12	19	12	19
36. Switchgear wt, # (estimate)	2	2	2	2
37. Integral Fuel Storage Tank, # (estimate)	77	77	77	77

TABLE 3-3 (Cont)

38. Integral H$_2$O Storage Tank, #	30	30	30	30
39. I.C. Engine Battery Pack, #	40	40	40	40
40. Σ 34 thru 39	254.7	261.7	254.7	261.7
41. Q rejection/Q power (Vickers report P-1565-389-1)	.628	.628	.628	.628
42. For Power @ 12, Q rej = 41 x 12, kw	3.27	3.39	3.27	3.39
43. Engine Discharge Press, Psia (Vickers report)	2.5	2.5	2.5	2.5
44. T sat @ 43, ^{o}R	594	594	594	594
45. Sp. Rad. Wt @ 44, α = .1, Po = .999 ($\underline{W}$ curve) #/kw	66	66	66	66
46. Rad. Wt = 45 x 42, #	216	224	216	224
47. 40 + 46, I.C. Engine Dry Wt, #	470.7	485.7	470.7	485.7
48.				
49. Stirling Engine System Weight, # (See F.C. Calc.)		661		661
50. Stirling Engine Radiator Weight, # α = .1, Po = .999		870		870
51. Stirling Engine Radiator Weight, # α = .3, Po = .999		2805		2805
52. System Plumbing Wt. = .10 (47 + 49 + 50), # α = .1		202		202
53. System Electrical Wt. = .05 (47 + 49 + 50), # α = .1		101		101
54. System Plumbing Wt. = .10 (47 + 49 + 50) α = .3		395		395
55. System Electrical Wt. = .05 (47 + 49 + 50) α = .3		101		101
56. Total Solar Engine Wt. = 49 + 50 + 52 + 53, # α = .1		1834		1834

TABLE 3-3 (Cont)

	Without Redundancy	With Redundancy 1 Extra Stirling 3 Extra I.C. Engines	Without Redundancy
57. Total Solar Engine Wt. = 49 + 51 + 54 + 55, # α = .3	3962		3962
58.			
59. Total System Dry Wt. = 56 + 47, # α = .1	2319.7		2319.7
60. Total System Dry Wt. = 57 + 47, # α = .3	4447.7		4447.7
61.			
62.			
63.			
64.			

CALCULATION OF SHELTER I.C. ENGINE & SOLAR STIRLING POWER SUPPLY TOTAL & SPECIFIC WEIGHTS

	Without Redundancy	With Redundancy 1 Extra Stirling 3 Extra I.C. Engines	Without Redundancy
65. Avg. kw-hr for 1 lunar day	108.7 x 3.75 = 2660 kw-R		
66. Avg. kw-hr for 2 lunar days	5320		
67. Avg. kw-hr for 3 lunar days	7980		
68. Avg. kw-hr for 5 lunar days	13300		
69. Avg. kw-hr for 10 lunar days	26600		
70. Avg. kw-hr for 20 lunar days	53200		
71. Avg. kw-hr for 24 lunar days	63800		
72. Total system wet wt for 1 lunar day, α = .1 = 21 + 59, #	4132	5573	4295
73. Total system wet wt for 1 lunar day, α = .3 = 21 + 60, #	6260	7701	6423
74. Total system wet wt for 2 lunar days, α = .1 = 22 + 59, #	5944	7385	6270
75. Total system wet wt for 2 lunar days, α = .3 = 22 + 60, #	8072	9513	8398
76. Total system wet wt for 3 lunar days, α = .1 = 23 + 59, #	7760	9201	8250

TABLE 3-3 (Cont)

77. Total system wet wt for 3 lunar days, $\alpha = .3 = 23 + 60$, #	9888	11329	10378
78. Total system wet wt for 5 lunar days, $\alpha = .1 = 24 + 59$, #	11380	12821	12200
79. Total system wet wt for 5 lunar days, $\alpha = .3 = 24 + 60$, #	13508	14949	14328
80. Total system wet wt for 10 lunar days, $\alpha = .1 = 25 + 59$, #	20440	21881	22070
81. Total system wet wt for 10 lunar days, $\alpha = .3 = 25 + 60$, #	22568	24009	24198
82. Total system wet wt for 20 lunar days, $\alpha = .1 = 26 + 59$, #	38560	40001	41820
83. Total system wet wt for 20 lunar days, $\alpha = .3 = 26 + 60$, #	40688	42129	43948
84. Total system wet wt for 24 lunar days, $\alpha = .1 = 27 + 59$, #	45820	47261	49720
85. Total system wet wt for 24 lunar days, $\alpha = .3 = 27 + 60$, #	47948	49389	51848
86. Total system Sp wt for 1 lunar day, $\alpha = .1 = 72/65$ #/kw-hr	1.55	2.095	1.615
87. Total system Sp wt for 1 lunar day, $\alpha = .3 = 73/65$ #/kw-hr	2.355	2.895	2.415
88. Total system Sp wt for 2 lunar days, $\alpha = .1 = 74/66$ #/kw-hr	1.116	1.387	1.178
89. Total system Sp wt for 2 lunar days, $\alpha = .3 = 75/66$ #/kw-hr	1.518	1.790	1.578
90. Total system Sp wt for 3 lunar days, $\alpha = .1 = 76/67$ #/kw-hr	.973	1.152	1.034
91. Total system Sp wt for 3 lunar days, $\alpha = .3 = 77/67$ #/kw-hr	1.239	1.415	1.30

TABLE 3-3 (Cont)

92. Total system Sp wt for 5 lunar days, $\alpha = .1 = 78/68$ #/kw-hr	.855	.964	.917
93. Total system Sp wt for 5 lunar days, $\alpha = .3 = 79/68$ #/kw-hr	1.017	1.123	1.078
94. Total system Sp wt for 10 lunar days, $\alpha = .1 = 80/69$ #/kw-hr	.768	.823	.830
95. Total system Sp wt for 10 lunar days, $\alpha = .3 = 81/69$ #/kw-hr	.848	.902	.910
96. Total system Sp wt for 20 lunar days, $\alpha = .1 = 82/70$ #/kw-hr	.724	.752	.786
97. Total system Sp wt for 20 lunar days, $\alpha = .3 = 83/70$ #/kw-hr	.766	.791	.827
98. Total system Sp wt for 24 lunar days, $\alpha = .1 = 84/71$ #/kw-hr	.718	.741	.780
99. Total system Sp wt for 24 lunar days, $\alpha = .3 = 85/71$ #/kw-hr	.753	.774	.813
100.			
101.			
102.			

D. PHOTOVOLTAIC POWER SYSTEMS FOR SPACE APPLICATIONS

Solar cells are solid state photovoltaic devices which convert a portion of the photon energy from the sun into electrical energy. Solar cells are feasible for use in lunar power supplies.

ENVIRONMENT

The environment on the moon is assumed to include: temperature extremes of $130^{\circ}C$ at noon to $-173^{\circ}C$ at night; meteoroids with an average mass of $6X\ 10^{-3}$ gm and an average diameter of 0.04 inches, traveling 10 meters per second with a density-frequency at one per 60 square meters per day; a gravity field equal to 0.16 that at the earth's surface; and equal light and dark periods of 336 hours duration. Figure 3-27 gives the moon surface-temperature profile which is being used in this study.

The moon is assumed to be free from high energy electrons and protons. If this is not the case, systems can overcome the radiation damage problem by either adding protective cover slips to solar cells or by adding to the solar array area and allowing cell degradation.

PHOTOVOLTAIC DEVICES

All types of photovoltaic devices have the characteristic of decreasing efficiency with increasing temperature within the range of temperatures encountered in this mission. Figure 3-28 shows a theoretical efficiency (based on the junction current being the ideal junction current) at various temperatures for different energy gap materials. Silicon and gallium arsenide experimental measurements agree with the gross theoretical behavior. Cadmium sulfide behaves anomalously as if it were a 1.1 ev band gap material. Figure 3-28 gives the basic material contenders for use in a lunar photovoltaic power supply.
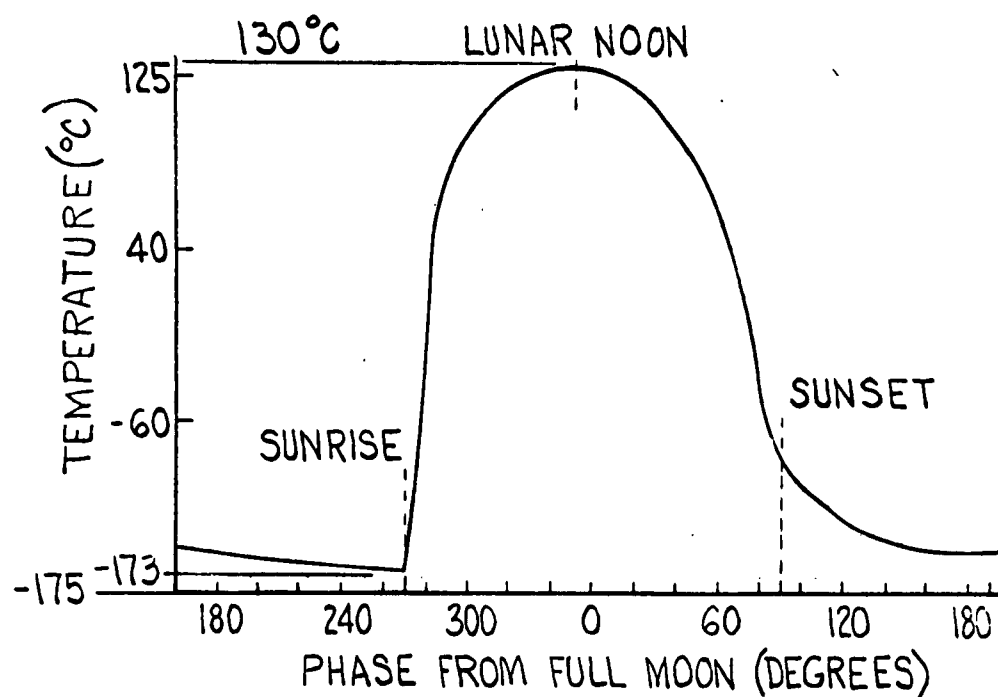
(REFERENCE 1)

Figure 3-27. Vibration of Temperature at Incidence of a Point at the Center of the Disc
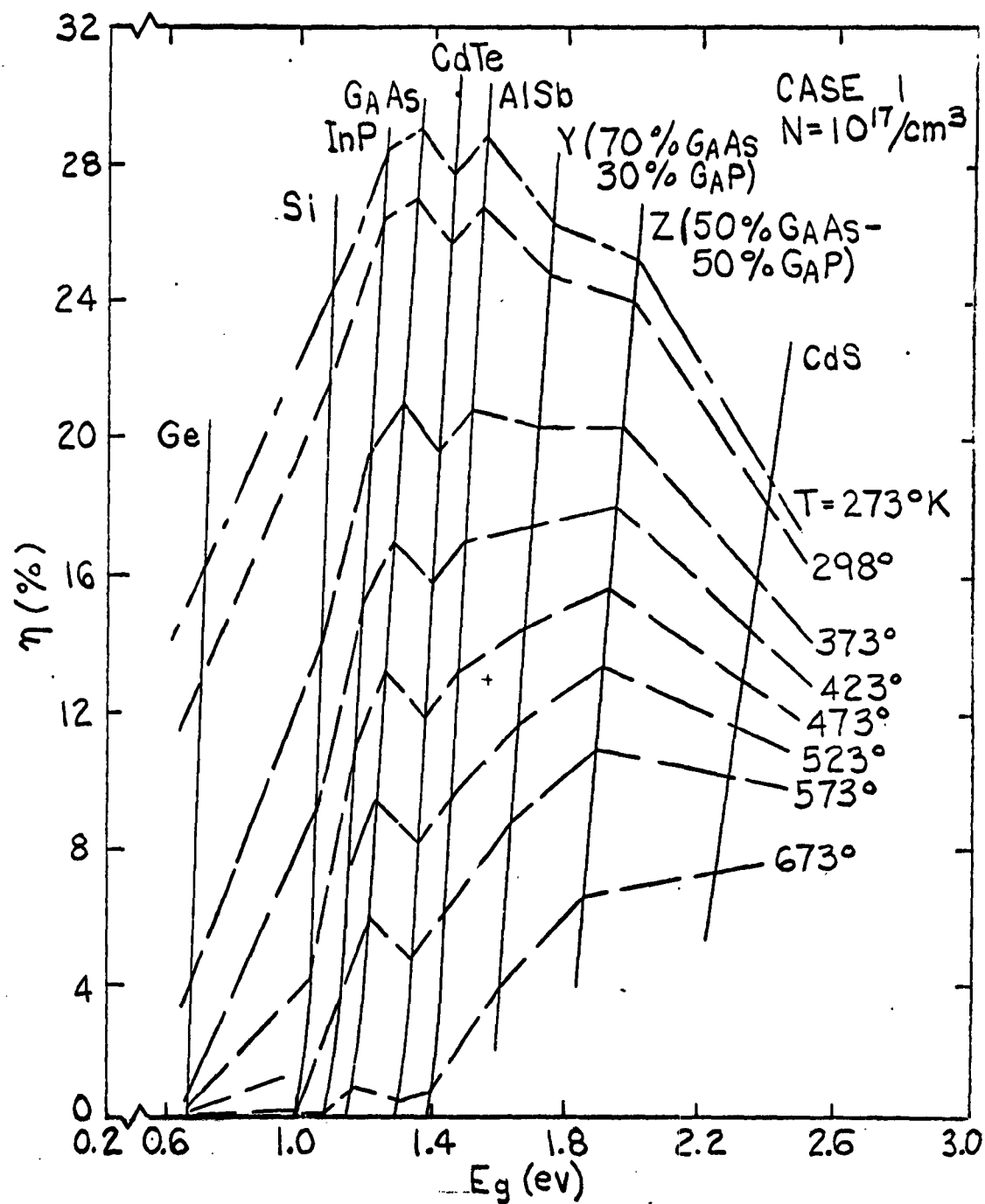
Figure 3-28. Efficiency Vs. Energy Gap (REFERENCE 2)

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APPROACH TO SOLAR CELL MATERIALS

The various materials in Figure 3-28 reflect a theoretical approach and assume some ideal conditions. The state-of-the-art is not equally advanced for all these materials. Silicon has received more development than the other types. Presently there are two types of solar cells available for use in space power-system application. These are the 1 cm by 2 cm silicon cells in the P on N and N on P dopings. Other types of cells are in various states of development. These include single crystal and/or thin film versions of silicon, gallium arsenide, cadmium sulfide, cadmium telluride, and gallium arsenide - gallium phosphide.

Silicon Solar Cells

Czochralski and float-zone type single crystals are the basic type material for solar cells presently in production. Solar cells from these crystals are normally 1 cm by 2 cm, but may be 2 cm by 2 cm. Presently in the advanced state of development prior to production is the silicon-web single crystal which is grown between dendrites. This material and automated semicontinuous processing have the potential of significantly reducing the cost of silicon cells. Polycrystalline thin film silicon cells are farther in the future than the silicon web but have a potential for further cost and weight reductions.

Two basic fabrication approaches are available to the silicon cell manufacturer. One approach is to dope the base material with one of the group-V elements thus giving it a surplus of electrons and making it N type. An element from group III may be diffused into the base material thus forming a P-N junction and a P on N type cell. The second approach is to interchange the use of valence groups of the doping elements and make an N on P type cell. The latter type cell has demonstrated superior resistance to damage from high energy protons and electrons. These types of cells are made by diffusing a top layer in a uniformly doped base material.

A further improvement in radiation resistant cells is to use the drift field technique where the doping material in the base region has a gradient from front to rear. This method is presently under development. Two other methods for making silicon cells are being studied. One is the use of an epitaxial growth top layer. The second is the use of ion bombardment to create a top layer.

Figure 3-29 shows the characteristics of efficiency change with cell temperature for conventional silicon solar cells. The two dashed lines represent the range for different cells. Figure 3-30 gives the spectral response for silicon cells at two different temperatures.

Gallium Arsenide

Gallium arsenide appears to be the material which could follow silicon in full scale production. The reduction in efficiency with increasing temperature is less than that for silicon, making it more suited for high temperature operation. Also, the resistance to damage by certain types of radiation is higher for gallium arsenide than for silicon cells. Gallium arsenide in the purity necessary for solar cell manufacture is basically expensive. Unless a thin film or epitaxial growth technique is perfected for this cell material, its price will always exceed that of silicon.

Figure 3-29 also shows the efficiency-temperature trend for gallium arsenide cells. As can be seen, an initially less efficient gallium arsenide cell can exceed the efficiency of a silicon cell as cell temperature is increased. Figure 3-31 shows the spectral response curve for a gallium arsenide solar cell.

Cadmium Sulfide

Cadmium sulfide thin film cells have been under development for some time and with various approaches. Two types are front wall and back wall, and are made by either vacuum evaporation or chemical spray deposition. Vacuum evaporation front wall cells are presently available in engineering evaluation quantities. The room temperature efficiency is substantially less than

3-105
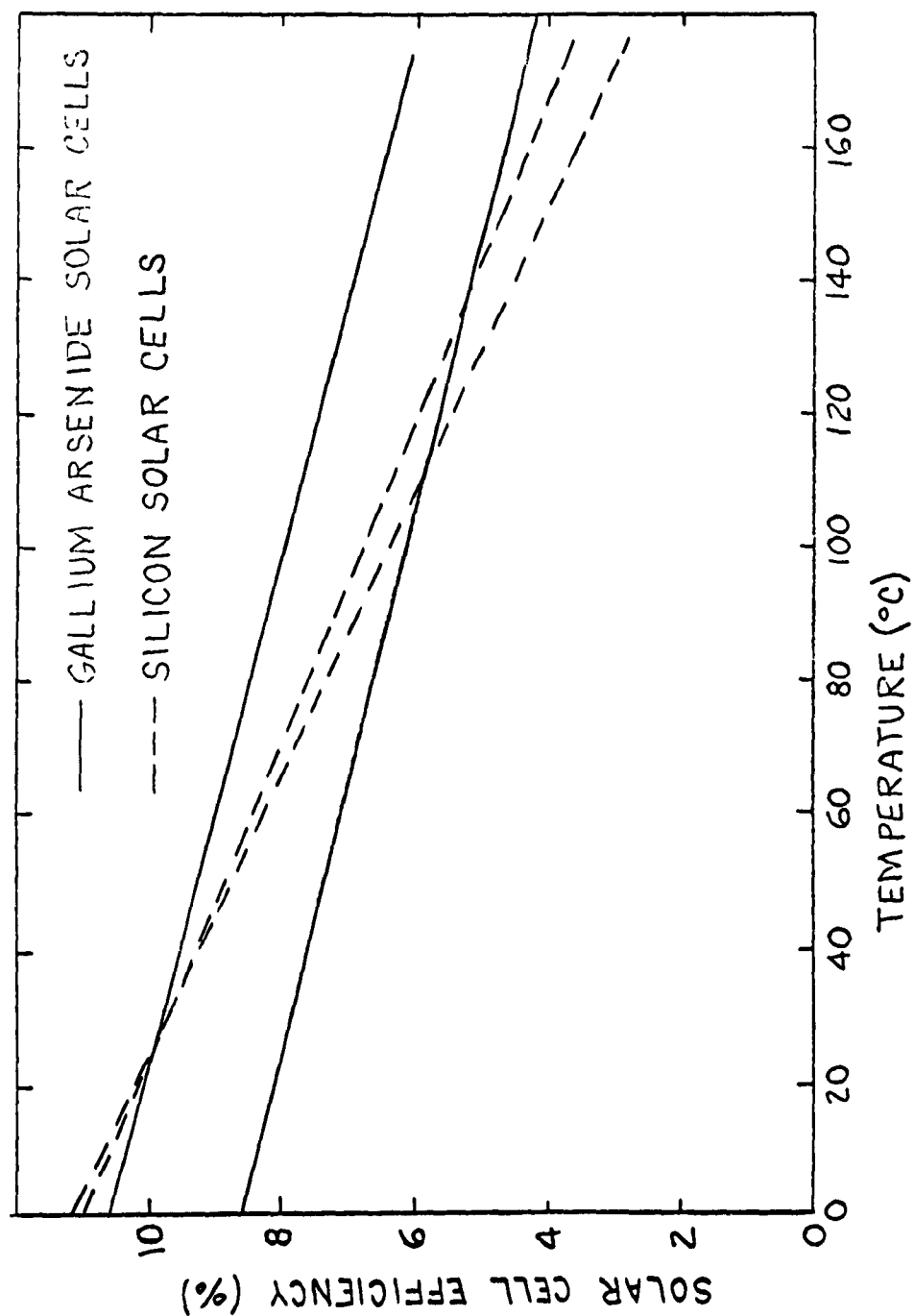
Figure 3-29. Temperature-Efficiency Characteristics for Solar Cells
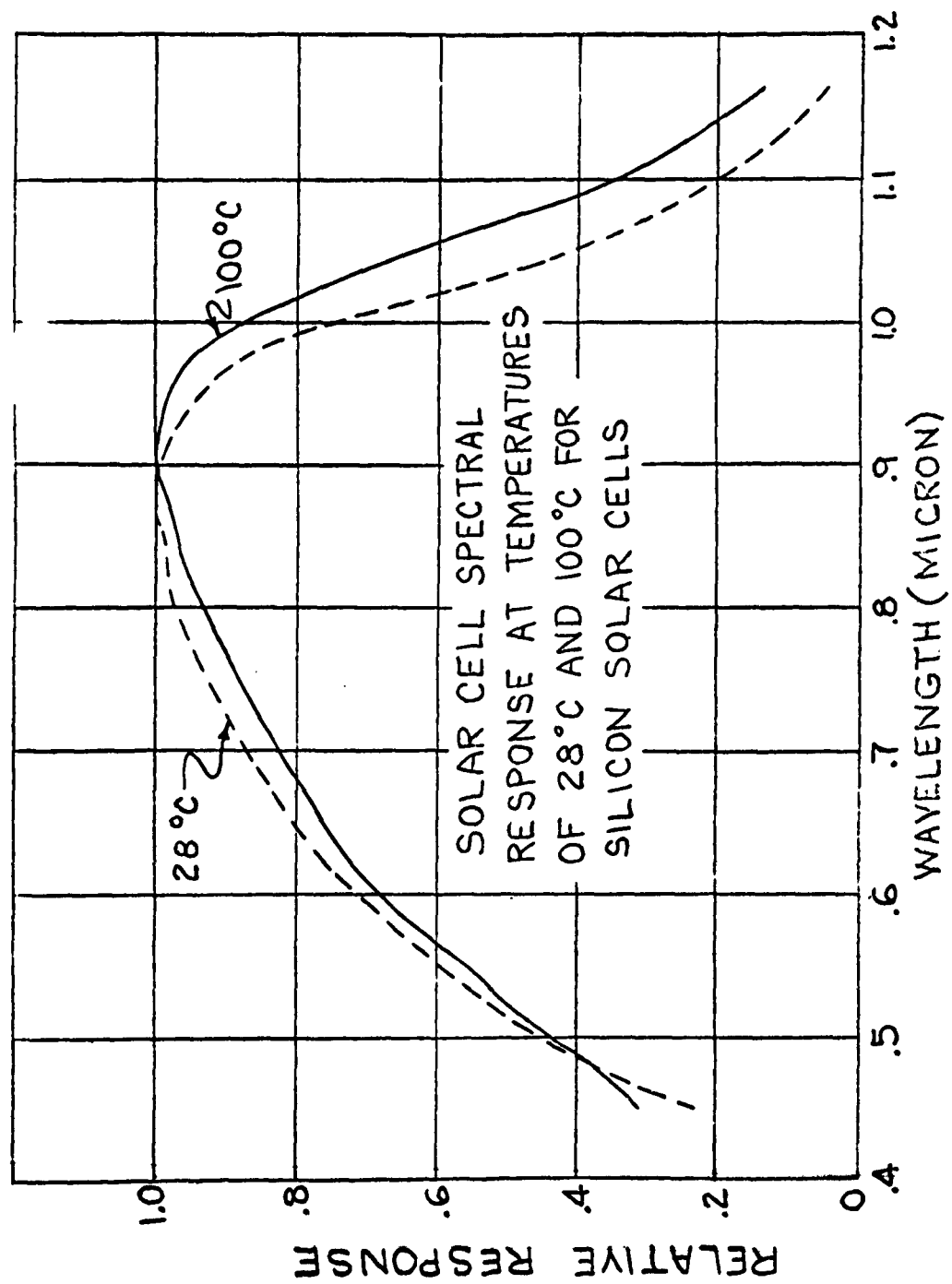
(REFERENCE 3)

Figure 3-30. Solar Cell Spectral Response

GₐAₛ CELL S47HC

Q (RESPONSE PER PHOTON)

λ (μ)

(REFERENCE 3)
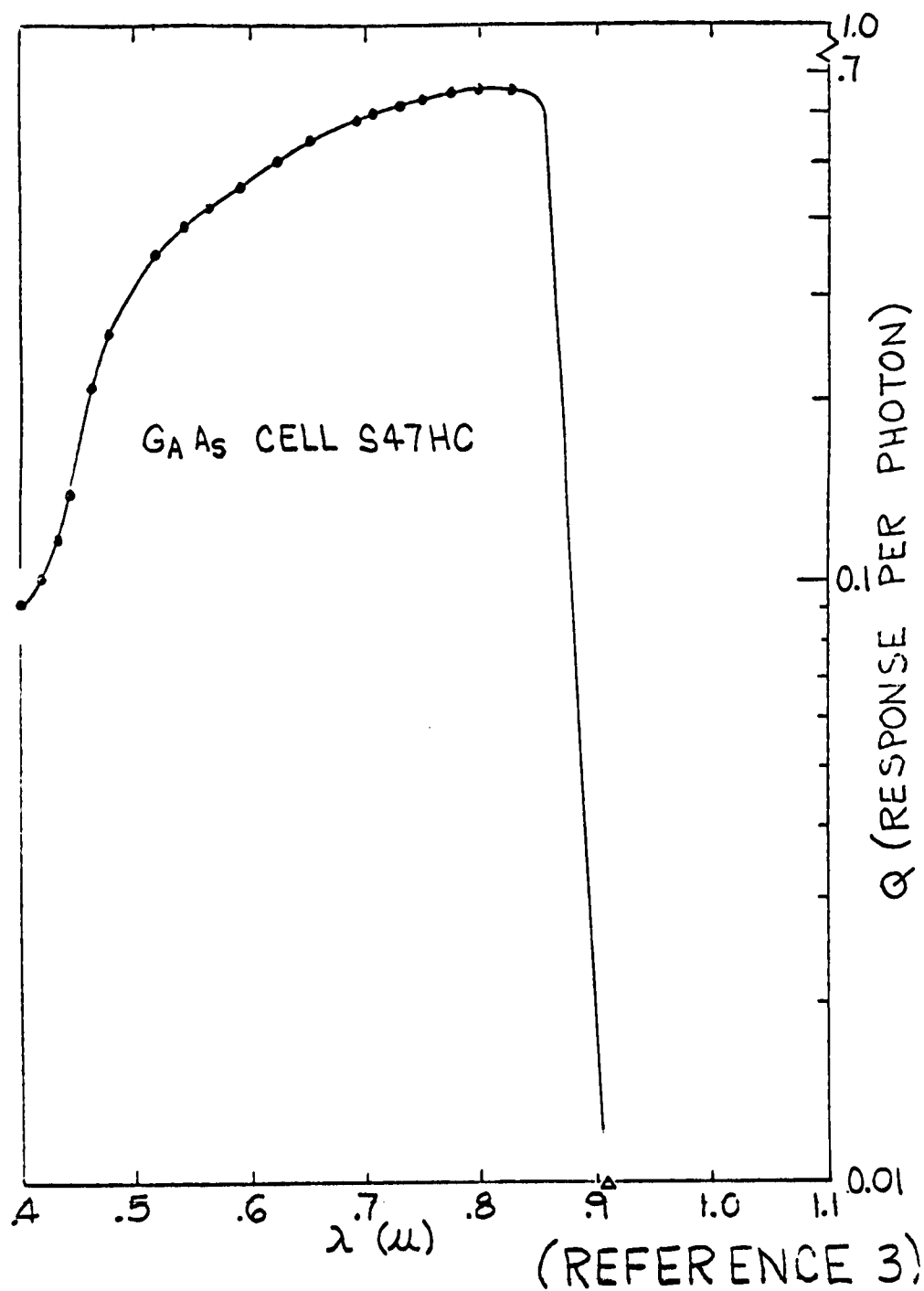
Figure 3-31. Spectral Response of GaAs Cell

3-108

for silicon cells, but the weight is less per unit area. This type cell has a potential for reducing the cost of space electric power and appears to have a good resistance to radiation damage.

Figure 3-32 shows the power and efficiency change with temperature for a panel of this type cell. Figure 3-33 shows the relative spectral response for front wall and back wall cells.

Cadmium Telluride

Cadmium telluride solar cells are being investigated as both thin film and single crystal devices. The emphasis has been on the thin film devices for production because of the possibility of high watts per dollar and per pound. The resistance to radiation damage appears to be good. This type cell is in the research laboratory state of development. Figure 3-34 shows the effect of efficiency, short circuit current and open circuit voltage on temperature. Figure 3-35 shows the relative response for two thin film cadmium sulfide cells. The dotted line in each case is for a cell with a more abrupt junction than for the cell represented by the solid line.

Gallium Arsenide - Gallium Phosphide

Gallium arsenide - gallium phosphide solar cells are in the laboratory state of development. These single-crystal, variable energy gap cells are the farthest from being available for application in a lunar electric system. Increase in temperature within the range of $25^{\circ}C$ to $175^{\circ}C$ causes efficiency to decrease at the rate of approximately 0.015 percent per degree centigrade (Reference 8). Figure 3-36 shows the spectral response characteristics of several gallium arsenide - gallium phosphide cells.

TEMPERATURE EFFECTS

Table 3-4 is a summary of the temperature effects on maximum power for the various types of solar cells discussed here.

POWER VS TEMP.
CdS ORBITAL TEST PANEL NO.1
AT 9.2 Ω LOAD

INTENSITY-CONSTANT
@ 140 mw/cm²

EFFICIENCY VS TEMP.
CdS ORBITAL TEST PANEL NO.1
AT 9.2 Ω LOAD
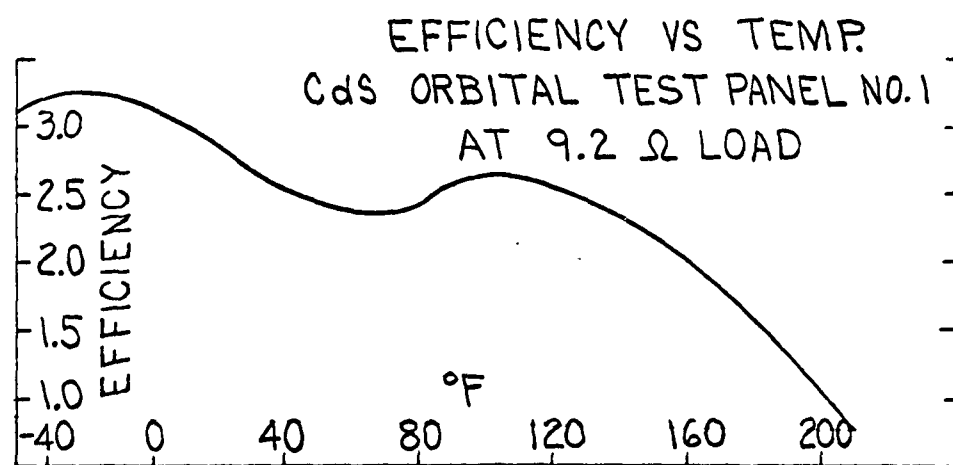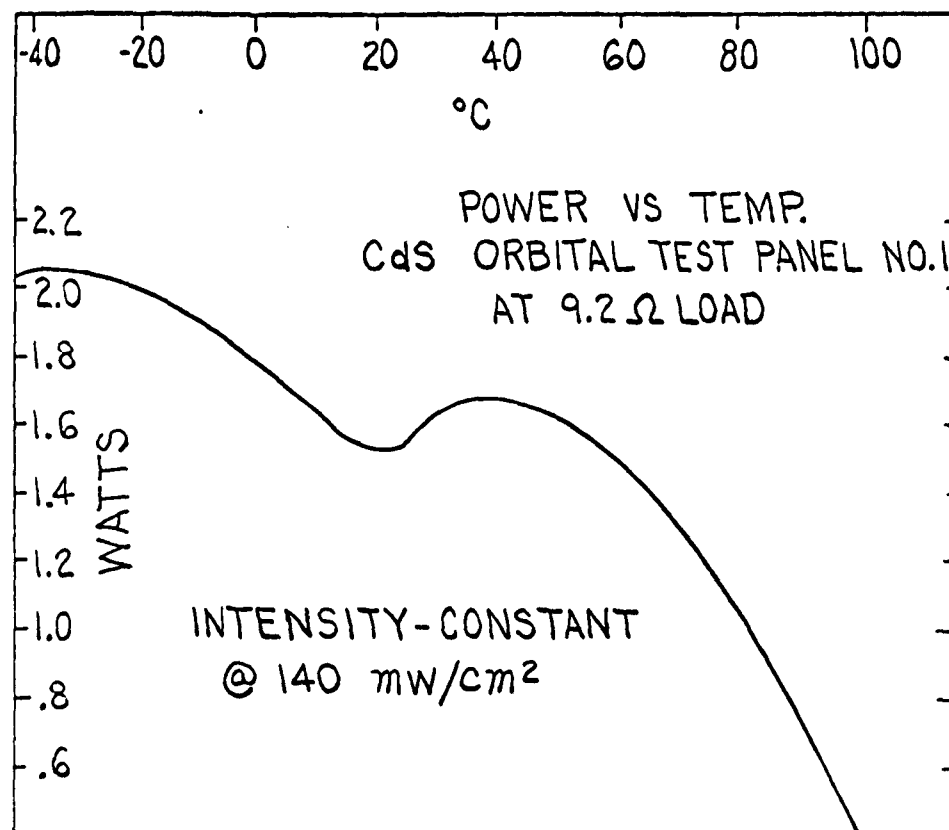
(REFERENCE 5)

Figure 3-32.   Power and Efficiency Vs. Temperature for CdS Cells

MONOCHROMATIC SPECTRAL RESPONSE CURVES

STABILIZED READINGS - NORMALIZED FOR LIGHT INPUT INTENSITY.

CELL NO. 700
ON GLASS SUBSTRATE

FRONT WALL CdS FILM CELL

REAR WALL CdS FILM CELL
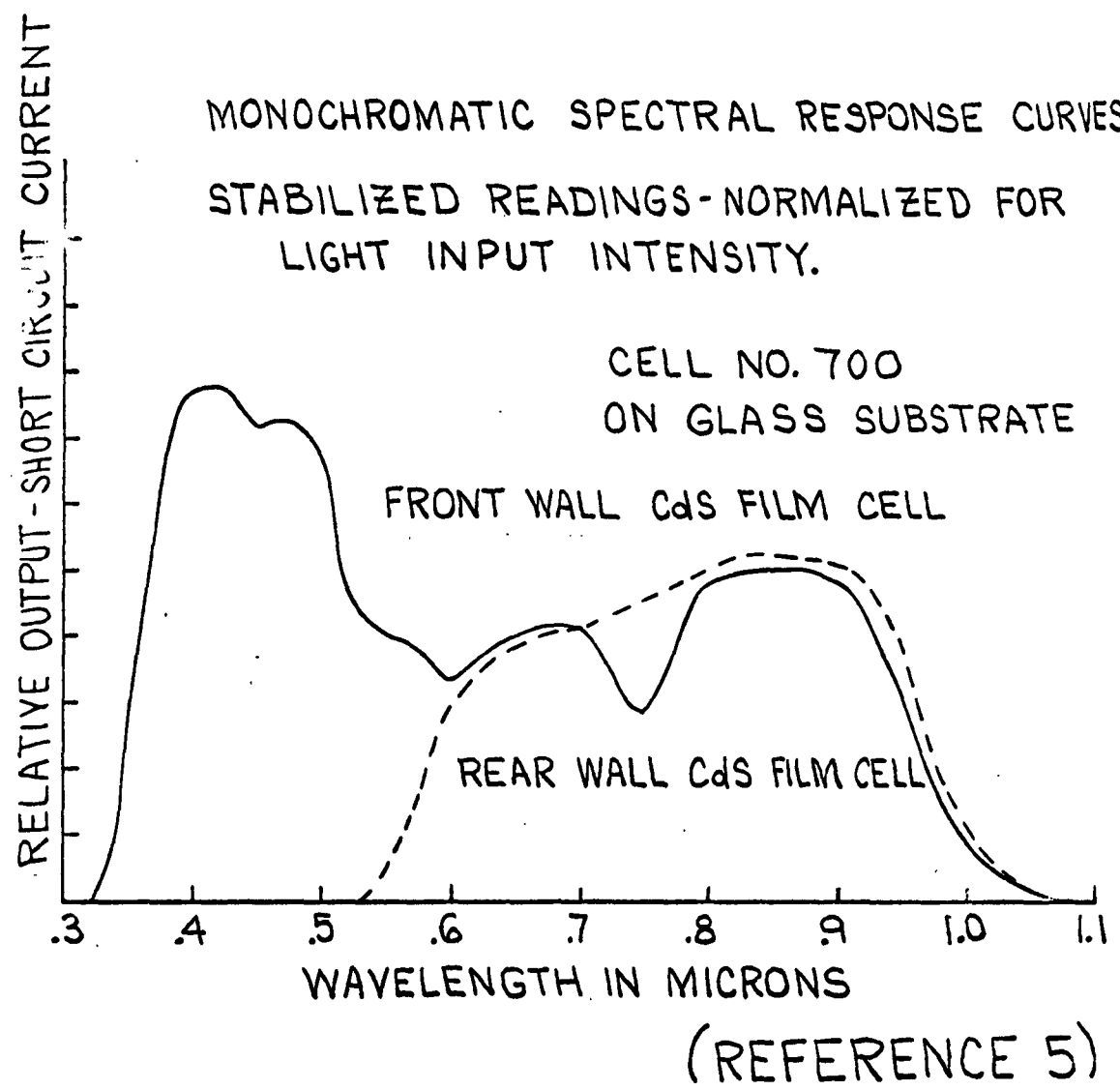
(REFERENCE 5)

Figure 3-33. Spectral Response of Rear Wall and Front Wall CdS Cells
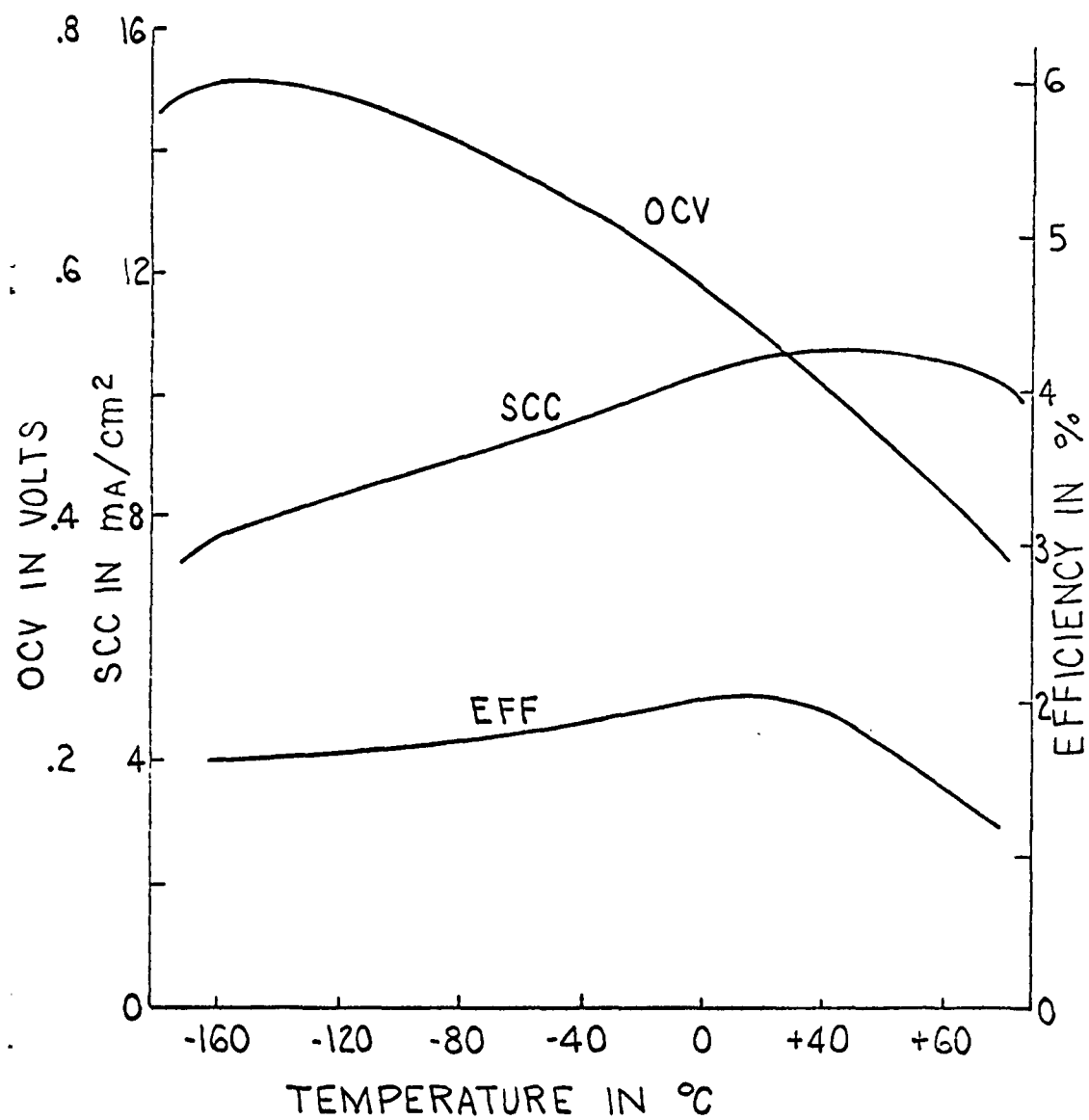
(REFERENCE 7)

Figure 3-34.
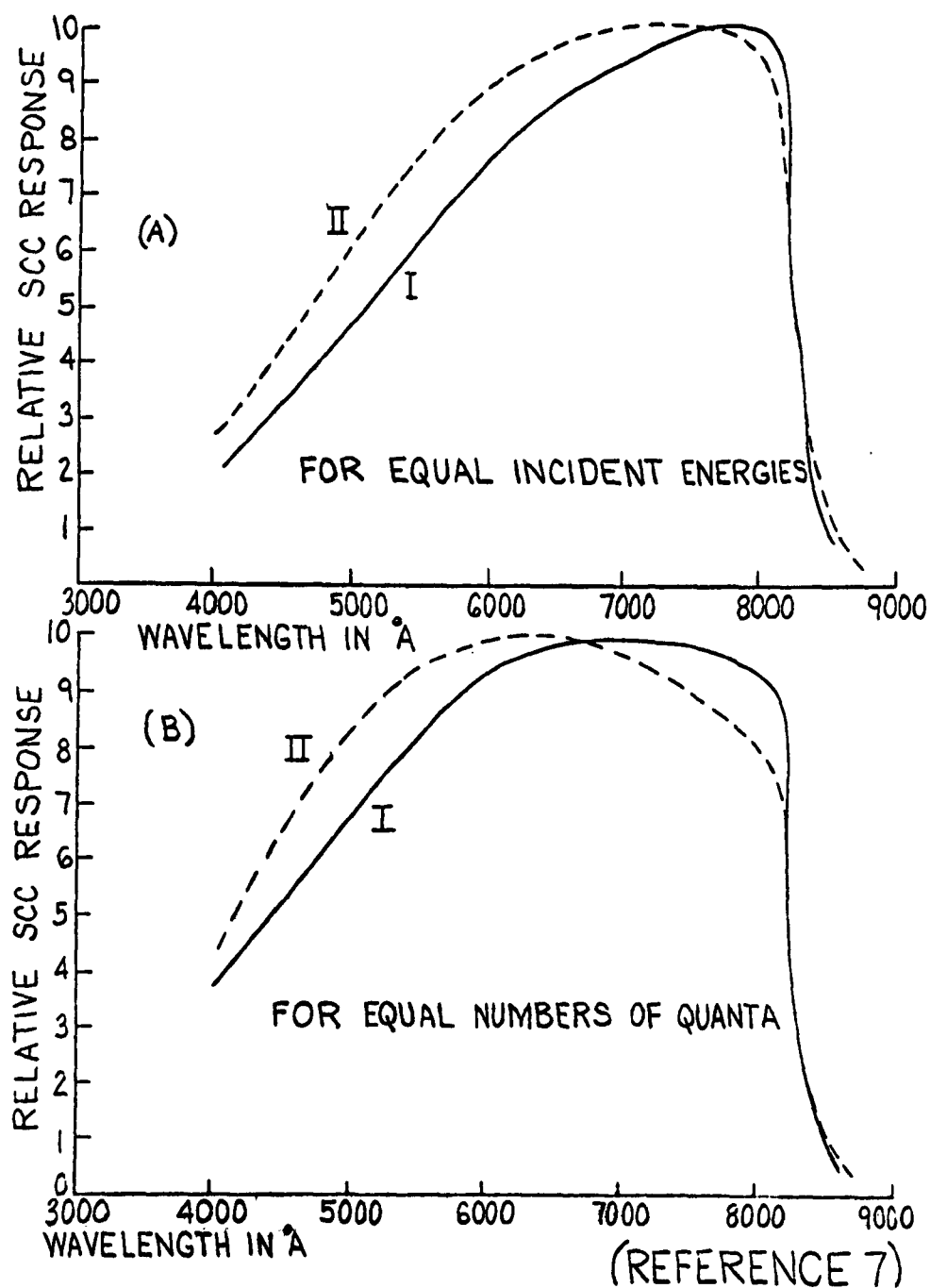
Figure 3-35.   Thin Film CdTe Solar Cell Response

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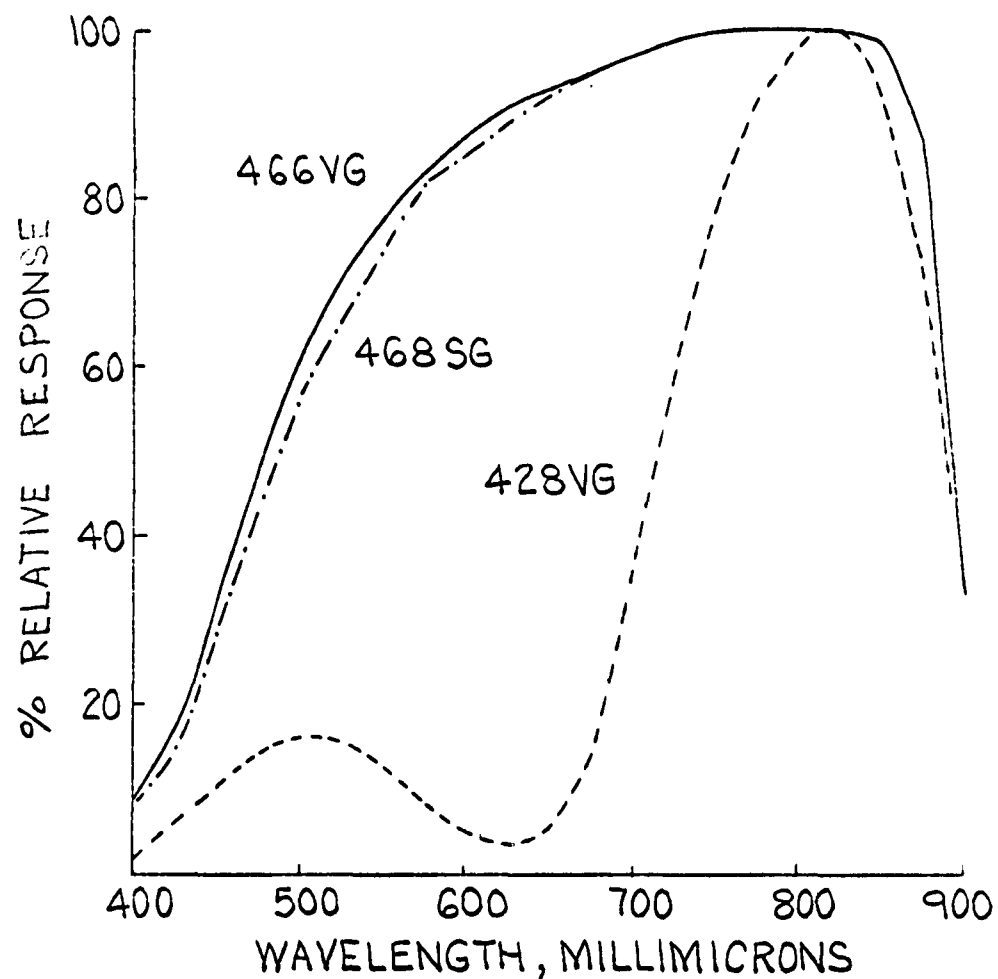
(REFERENCE 9)

Figure 3-36. Relative Response of Several Gallium Arsenide - Gallium Phosphide Solar Cells as a Function of Wavelength

TABLE 3-4

Solar Cells by Material Type	Change in Relative Efficiency with Increasing Temperature (25°C to 150°C)
Silicon	-0.043 to -0.045%/°C[3]
Gallium Arsenide	-0.024[3] to -0.027%/°C[8]
Cadmium Sulfide	-0.040 to -0.045%/°C[6]
Cadmium Telluride	-0.020%/°C[6]
Gallium Arsenide	-0.015%/°C[8]

Panel Thermal Considerations

Table 3-4 and the various temperature-efficiency curves show the degrading effect of increasing temperature, and Figure 3-27 shows the temperature of the moon's surface.

There are several approaches to reducing the solar cell operating temperature. The most desirable way to reduce solar cell temperature is to convert the maximum amount of the incident energy and conduct it away electrically. Solar cell efficiency determines the limits in this area and the state of development determines the values which may be used on a system design. This area is dependent on the solar cell supply industry.

A second way to reduce solar cell temperature is to reflect (with either specular reflectors or interference filters) the energy which the solar cells can not use or can use to little advantage. This area is dependent on the state of development of selective reflectors and filters.

A third way to reduce cell temperature is to dissipate the absorbed thermal energy with maximum effectiveness. High emittance coatings are presently available and others are being developed. The solar cell power system designer definitely has an opportunity to optimize a design by selecting components and applying them in clearer designs.

Figure 3-37 gives a plot of the energy at the moon's surface as a function of wavelength. The area under this curve is approximately 130 watts per square foot. A comparison of information on solar cell spectral response shows that no solar cells considered here utilize all of the energy available. Silicon cells operate within the 0.4 micron to 1.1 micron range which represents about 65 percent of the sun's energy. Gallium arsenide, cadmium telluride, and gallium arsenide - gallium phosphide operate in the range of 0.4 micron to 0.9 micron which covers about 55 percent of the sun's energy. Reducing the thermal load to a solar cell without reducing the energy which it can use is one way of increasing the cells output.

The emittance of P on N silicon cells both with and without a glass cover slip is given in Figure 3-38. A significant increase in emittance can be gained with the addition of a cover slip. High emittance, of course, helps reduce solar cell temperature.

SOLAR CELL PANEL THERMAL CHARACTERISTICS

A reference solar cell panel thermal design is shown in Figure 3-39. The location of this lunar power supply is at the moon's equator therefore only one axis of orientation is necessary. The panel pivots about one central axis and is oriented normal to the sun during the lunar day.

Two approaches were taken in thermal design analysis. One was to have the moon's surface exposed so that the panel received heat from both the sun and the moon and rejecting it to the moon and deep space. The second approach uses an aluminum foil shield on the lunar surface in the vicinity of the solar panel.

A thermal balance for each approach was investigated with the aid of a computer using the following assumptions where applicable:

1. Solar flux is a constant 130 watts per square foot.

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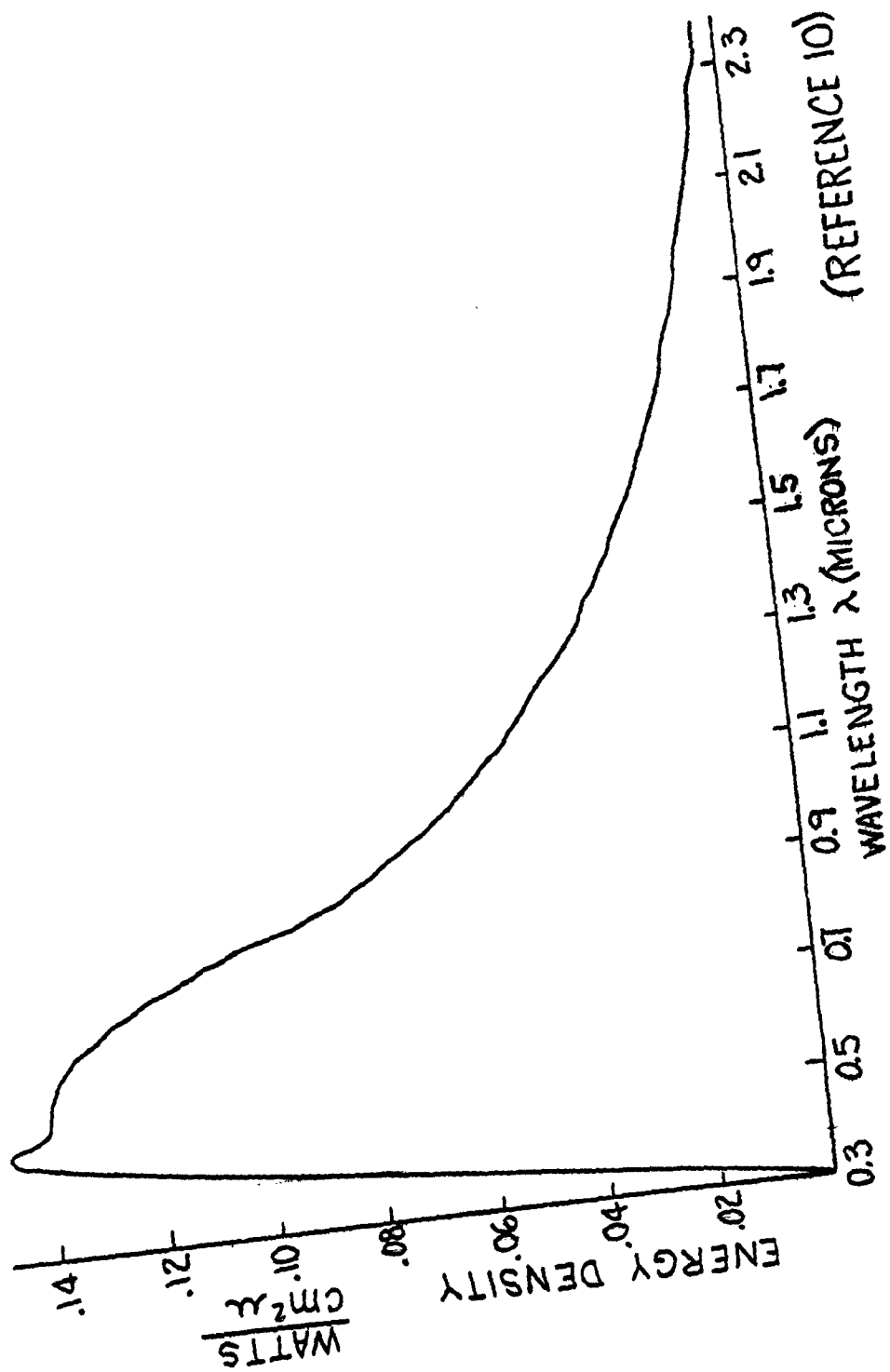
Figure 3-37. Energy Distribution of Solar Irradiation near Earth

3-117
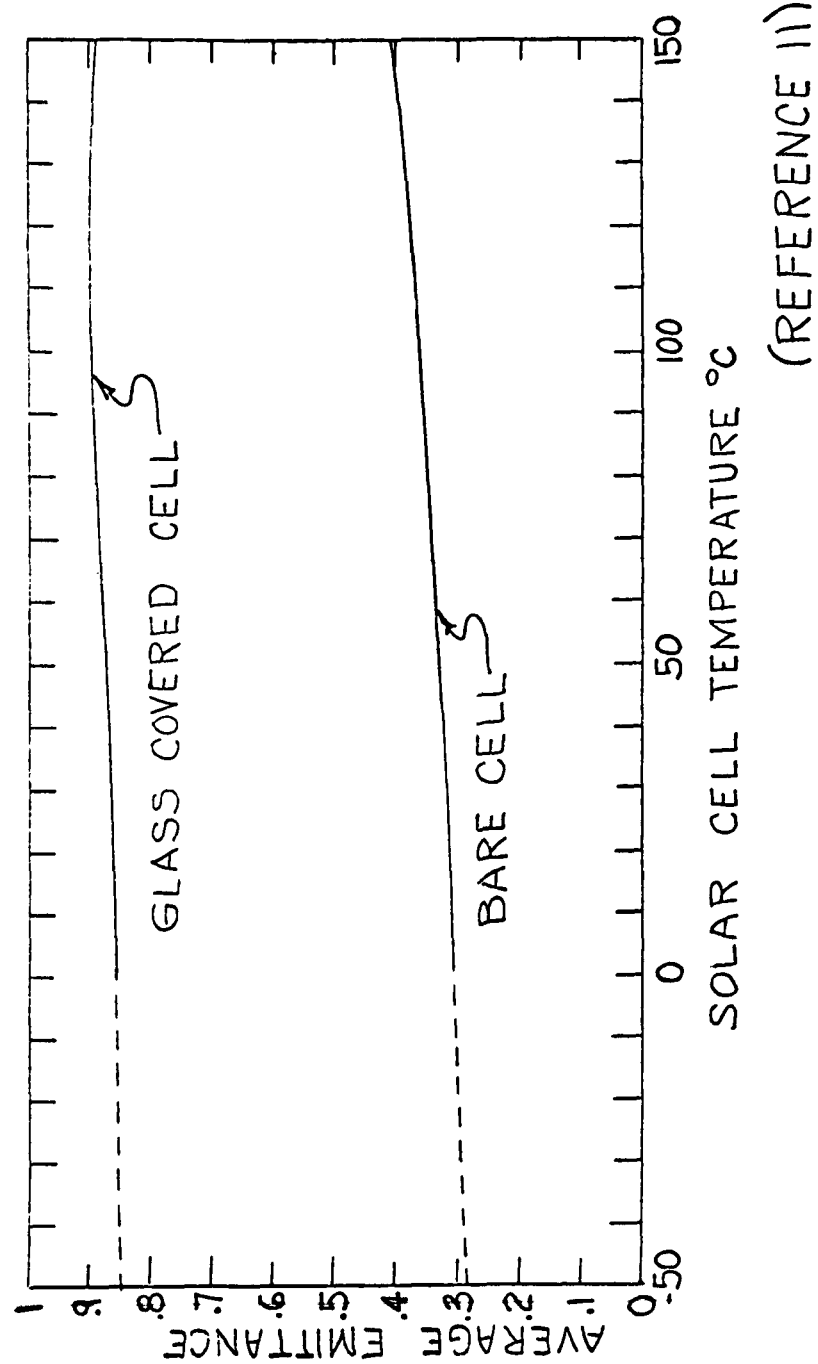
Figure 3-38. Bare and Glass-Covered Cell Emittance
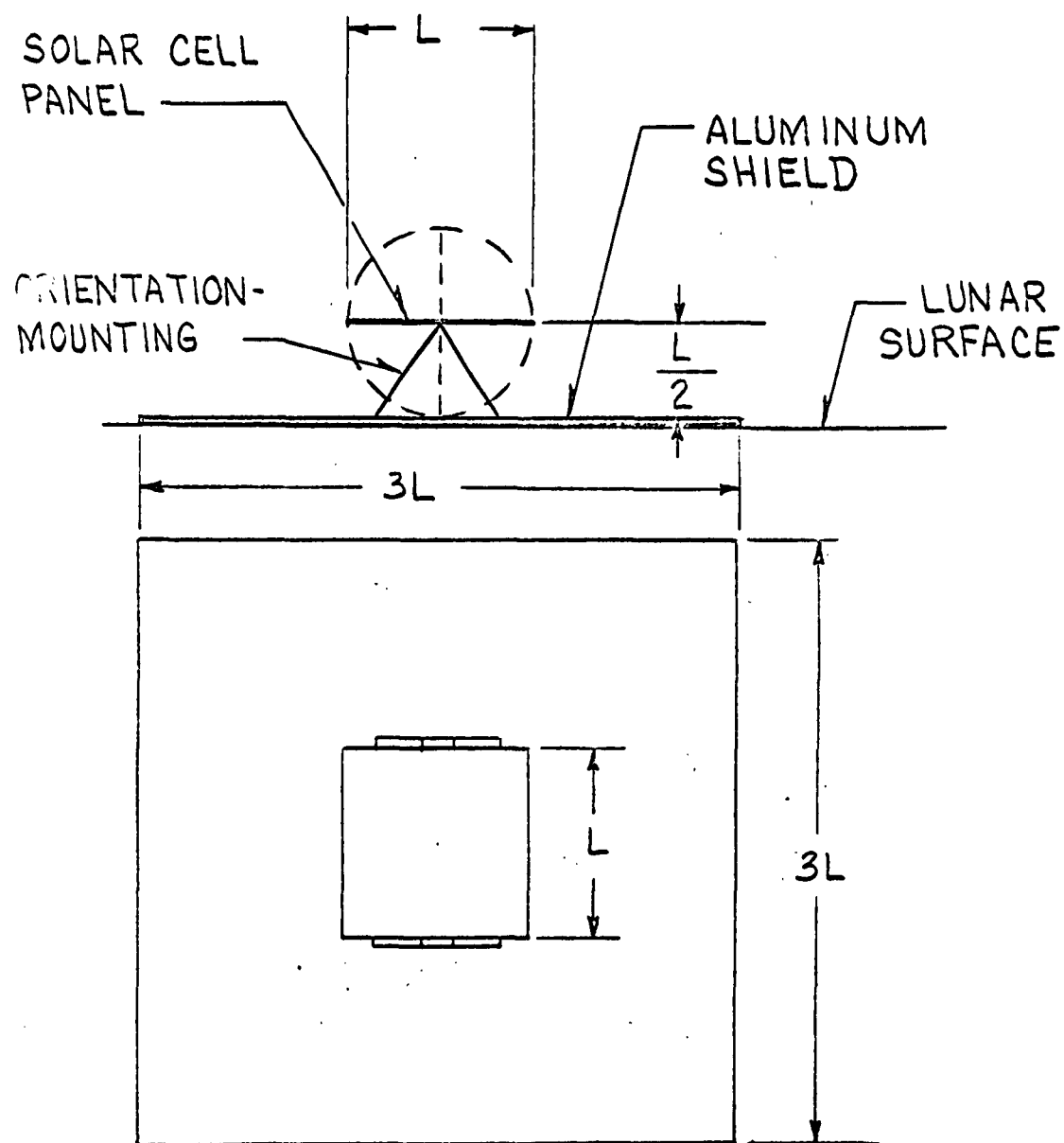
Figure 3-39.

2. The visible portion of the lunar surface is uniform in temperature and varies according to Figure 3-27 except where the surface is covered by aluminum. The thermal conductivity of the moon was assumed to be 3.3×10^{-5} cal/sec/cm/°C[12], therefore the aluminum surface absorptivity-emissivity ratio (0.5) is the factor governing the shield's temperature with the sun as the only heat source. The moon is an insulator which neither accepts or contributes to the shield's temperature.

3. ne lunar surface is an infinite flat plane.

4 The panel surface is always normal to the sun.

The thermal balance for the first approach results in the following equation where T is the panel equilibrium temperature in degrees Rankine.

$$T_1 = \left[\frac{3130 \times 10^8 \times BETA + (FFM \times EF + FBM \times EB) \, TM^4}{EF \times EB} \right]^{1/4}$$

where,

BETA = the solar absorptivity of the panel minus the solar cell efficiency

FFM = the view factor from the front of the panel to the moon

FBM = the view factor from the back of the panel to the moon

EF = the emissivity of the front of the panel

EB = the emissivity of the back of the panel

TM = temperature of the moon in degrees Rankine (from Figure 1)

The thermal balance for the second approach (with the shield) results in the following equation where T_2 is the panel equilibrium temperature in degrees Rankine.

$$T_2 = \left[\frac{3130 \times 10^8 \times BETA + (FFPM \times EF + FBPM) \, TM^4 + (FFA \times EF + FBA \times EB) \times TA^4}{EF + EB} \right]^{1/4}$$

Some of the variables are the same as used in the previous equation.

FFPM = the view factor from the front of the panel to the moon

FBPM = the view factor from the back of the panel to the moon

FFA = the view factor from the front of the panel to the aluminum

FBA = the view factor from the back of the panel to the aluminum

TA = the temperature of the aluminum in degrees Rankine

It should be noted that:

FFPM $\neq$ FFM

and

FBPM $\neq$ RBM

but

FFPM = 1 - FFS - FFA

and

FFBM = 1 - FBS - FBA

where,

FFS and FBS are the view factors from the front and the back of the panel to space, respectively.

Figure 3-40 gives a plot of panel temperature as a function of the time of the lunar day for various values of BETA. The value of BETA may be decreased by allowing only the usable solar energy to be absorbed by the panel, and by increasing the conversion efficiency of the solar cells. The effect of the shield is to decrease the panel temperature thereby increasing the efficiency of the solar cells.

Figure 3-41 shows how the maximum power efficiency varies with time of day for a range of solar cell characteristics (η_T = 0.015 to 0.045 percent/^{O}C) and for solar cell panel BETA of 0.60, 0.80 and 0.90. The term η_T is the negative coefficient of efficiency change with increasing temperature in the range of interest here and is shown in Table 3-4. This plot is for solar panels with the aluminum shield geometry as shown in Figure 3-39.

3-121
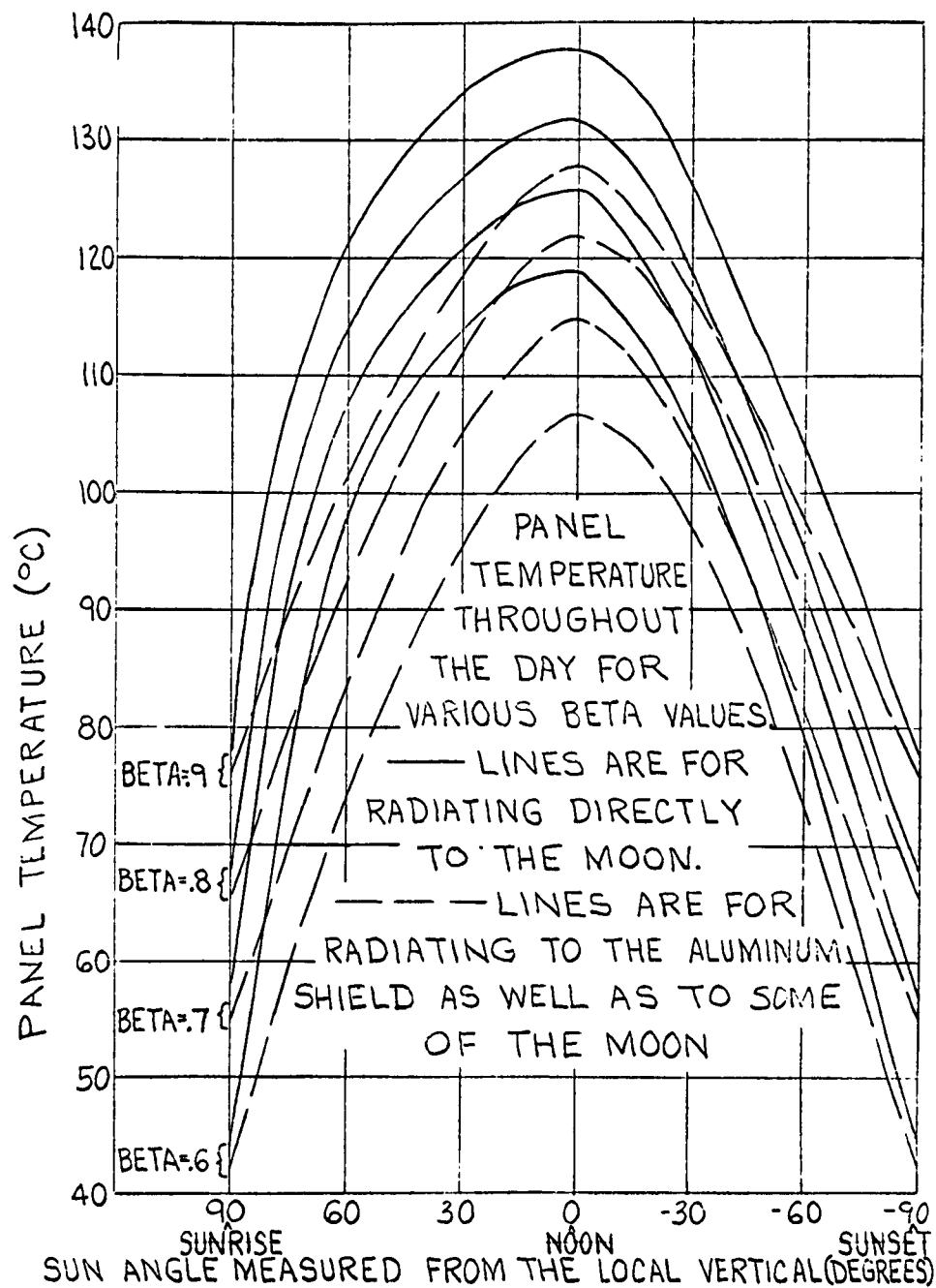
Figure 3-40.   Panel Temp. Throughout the Day for Various Beta Values
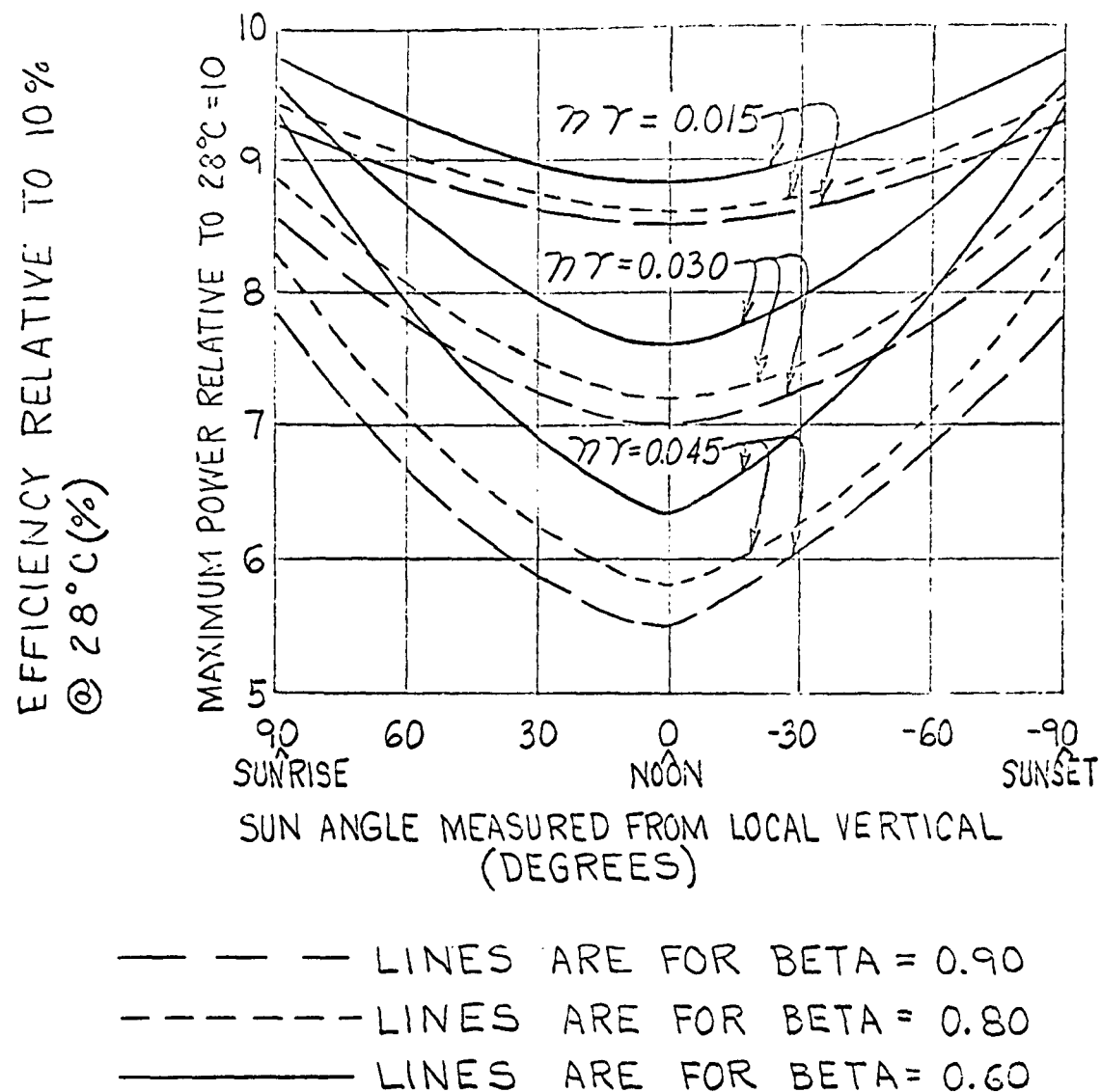
Figure 3-41. Relative Efficiency and Maximum Power as Function of Time of Day

REFERENCE CONCEPTS

Silicon Solar Cells

A set of reference concept designs for silicon solar cell systems of various power levels have been created using parameters that coincide with the present state-of-the-art. These concept designs represent a realistic approach which may be used as a basis for evaluating future concepts. The reference concept is based on the following assumptions.

1. The solar power density normally incident on the panel is 130 watts per square foot. The stacking factor (solar cell total area per square foot of panel) is 0.90. The ratio of solar cell active area to solar cell total area is 0.90. Therefore the intercepted energy available for conversion is 105.3 watts per square foot.

2. The solar cells are 1 cm x 2 cm silicon (here $\eta\tau$ = 0.045).

3. The average efficiency of the solar cells complete with cover slips and mounted in the array is 9 percent at 28°C in space sunlight. (This represents an efficiency between 11 and 12 percent for bare cells tested under 2800°K tungsten light.)

4. Solar cells are equipped with glass cover slips 0.006 in. thick.

5. The cover slips have an ultraviolet interference filter which reflects 10 percent of the energy incident on the cover slip.

6. The average emittance of the front of the panel is 0.85.

7. The average emittance of the back of the panel is 0.95.

8. The panel and mounting geometry is as shown in Figure 3-39, with a shield having an absorptivity to emissivity ratio of 0.5.

Figure 3-42 shows the maximum power output per square foot throughout the day, with and without the shield. Peak power is obtained at sunrise and sunset when panel temperatures are low. The minimum power occurs at noon and is 5.21 watts per square foot with the aluminum shield and 4.78 watts per square foot without. The average power throughout the day is
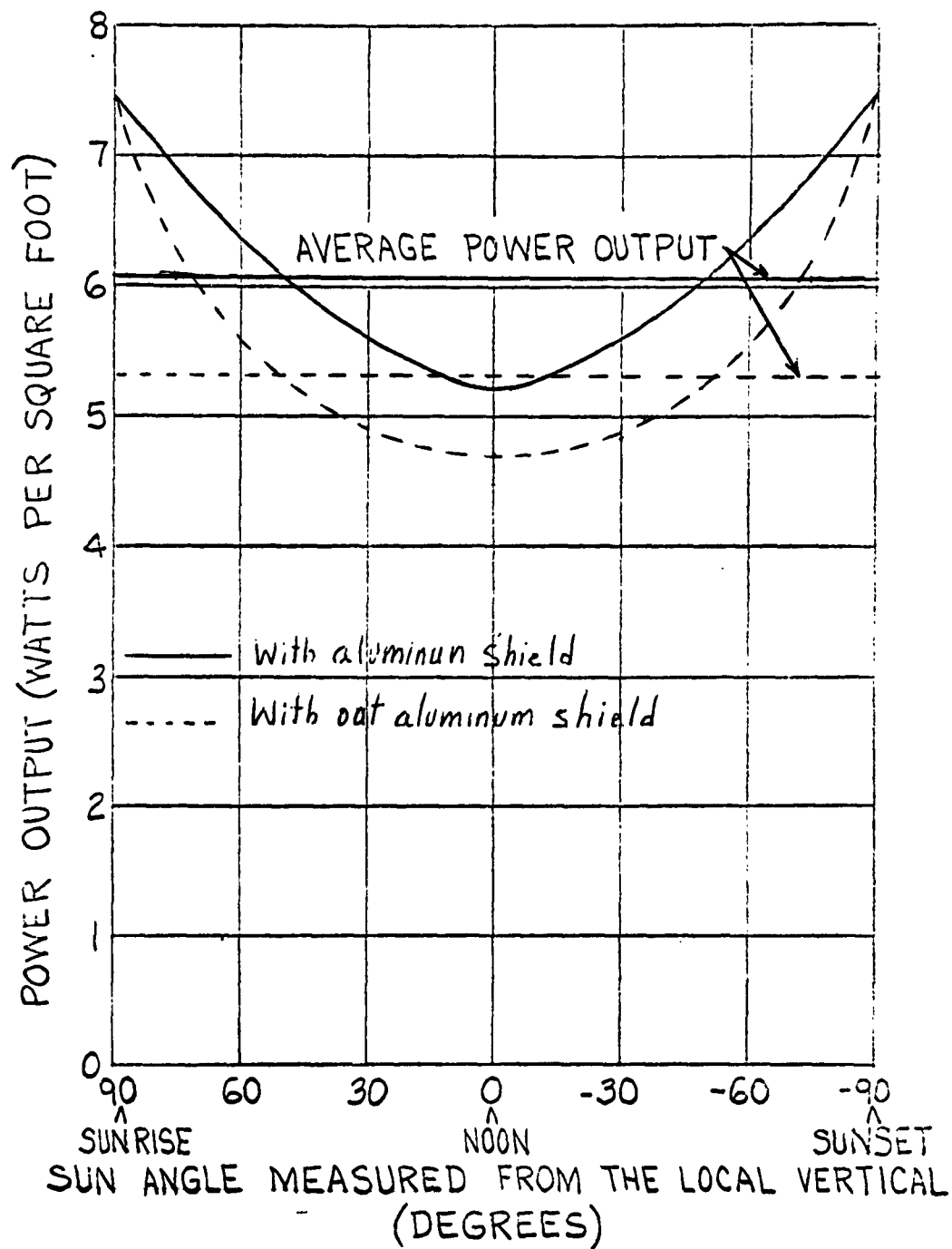
Figure 3-42. Reference Concept Power Throughout the Day

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6.06 and 5.30 watts per square foot for the with and without shield designs respectively. Maximum power voltage approximately follows the maximum power curve. This concept is based on information from Table 3-4 and Figures 3-40 and 3-41, with BETA equal to 0.85.

Power from this array is directly proportional to the array area if electrical transmission losses are neglected. Transmission losses are directly proportional to distance and inversely proportional to conductor area. Each array of a modular power system can be similar in design and wiring if arrays are all an equal distance from power conditioning and/or utilization equipment. Two approaches are available if this is not the case: use heavier conductors so the transmission losses are equal from all arrays; and add extra solar cells in each series path to furnish power lost in longer transmission. The following reference concept designs assume that power output is directly proportional to area and that power transmission losses are equalized by power conductors. Table 3-5 gives a breakdown description of several sizes of silicon solar cell arrays.

Figure 3-43 shows a noon specific power range as a function of array power and is based on information from Table 3-5. The small arrays ($\sim$ 100 watts) have a relatively high specific power. This size array may be impractical for use in larger power (kilowatts) systems because of the many modules involved. Large power systems may be assembled from modules of a selected size with the module size determining the system specific power.

The assumptions used in the Table 3-5 reference conceptual designs give the expected trend of decreasing specific power (watts per pound) with increasing array size. Alternate approaches may be used but the specific power decrease will persist.

A V-ridge concentrator could be used. Here the number of solar cells would be reduced to half and the power output would be reduced to
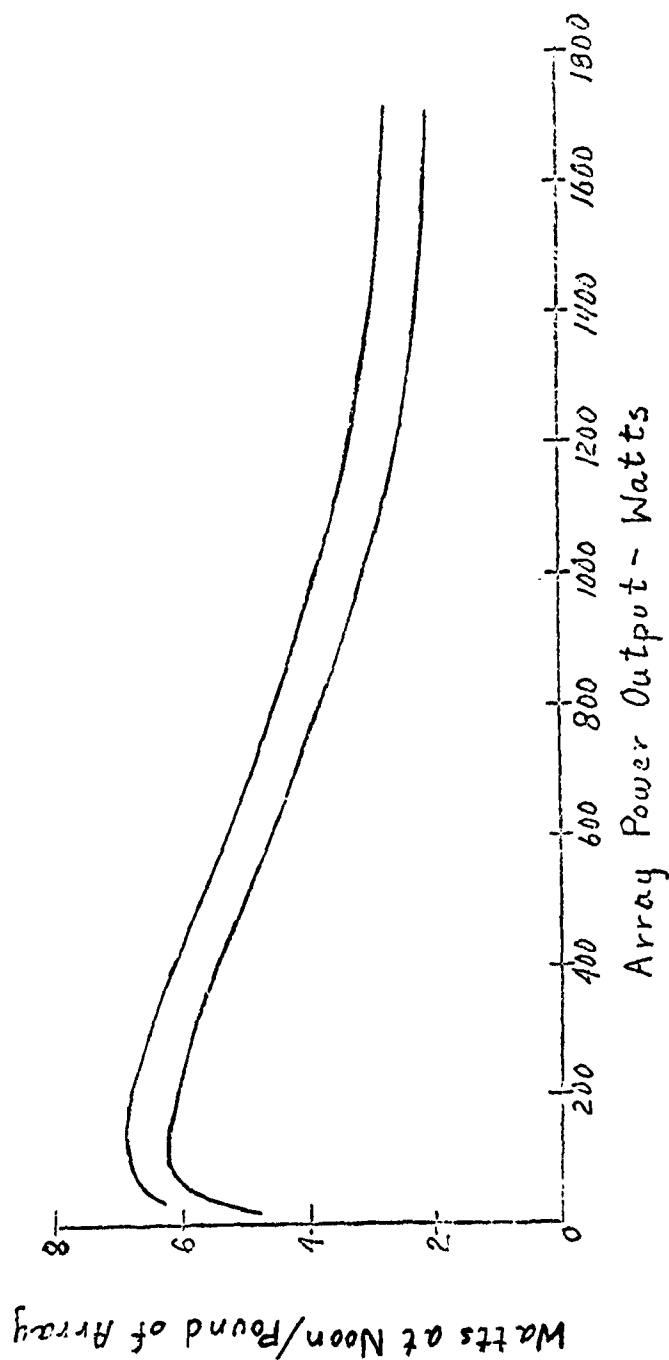
Figure 3-43

Array Power Output – Watts

Watts at Noon/Pound of Array

TABLE 3-5 (Cont)

	Solar Array Area (Ft)^2				
	16.5	100	165	250	330
Watts Avg. Power/Array wt. with 0.001" shield (watts/lb)	7.25	5.72	4.54	3.50	2.92
Watts Avg. Power/Array wt. with 0.0005" shield (watts/lb)	7.87	6.22	4.76	3.65	3.03
Watts Noon Power/Array wt. with 0.001" shield (watts/lb)	6.22	5.05	3.89	3.04	2.50
Watts Noon Power/Array wt. with 0.0005" shield (watts/lb)	6.75	5.39	4.10	3.16	2.58
Watts Noon Power/Array wt. (no shield) (watts/lb)	6.8	5.32	3.99	3.04	2.45

No. of cells per square foot (1 x 2 cm)   420

Weight of cells, cover slips and adhesive $0.3 \dfrac{lbs}{(ft)^2}$

*   The weight per unit area here varies as the total weight of the solar cells times 10^{-2} (with the exception of the 100 watt design).

**   The weight of the frame is equal to one half the panel weight.

*** The two A frames weigh in proportion to the array size and weight.

**** These weights are based on an aluminum shield thickness of 0.001" (Household foil has a thickness of 0.0007 inches).

approximately 4.74 watts per square foot at noon for the same cells with the shield configuration (a reduction of approximately 9 percent). There is a possibility of reducing panel weight with this configuration.

These reference designs are somewhat conservative. The following are areas in which improvements may be made. The stacking factor and cell area utilization factors could be improved, thus increasing watts per pound. A different type cell with a higher efficiency and/or lower decrease in efficiency with increasing temperature ($\eta\tau$) may someday be available. An

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effective blue-red filter which would reduce the BETA value may be developed. A shield with a lower ratio of absorptivity to emissivity may be used.

Cadmium Sulfide Solar Cells

The next type cell to be available in quantity will probably be the cadmium sulfide front wall thin film cell. Messrs. James Schaeffer, Herbert Lawrence, and Bell Managan of Harshaw Chemical Company (Reference 13) furnished the following information, from which conceptual design information on cadmium sulfide solar cells was created.

At room temperature, -100 mw per cm^2, and using a 3400^OK tungsten light, the efficiency for 3 inch by 3 inch cadmium sulfide solar cells is 3 percent. The maximum efficiency for 1 inch by 3 inch cells is 5.2 percent. The maximum efficiency for 6 inch by 6 inch cells is 3 percent. From this information, it seems reasonable to base designs on an average efficiency of 3 percent for 3 inch by 3 inch cells and a future average efficiency of 5.2 percent for 6 inch by 6 inch cells.

Three points were obtained from an efficiency vs. temperature curve with the same type discontinuity as in Figure 3-32. The 30^OC efficiency was 2.56 percent. The efficiency temperature curve was a straight line between 1.86 percent at 90^OC and 0.69 percent at 140^OC. In the reference designs the efficiency axis was translated so that the 30^OC temperature efficiencies were 3 percent and 5 percent.

A range of empirical factors to convert the power output received from tungsten light rating to space light has been determined. This range is from 1.2 to 2.0. A value of 1.4 was used for a present day cell design and 1.8 for a future cell design. A nominally 6 inch by 6 inch cell has an active area of 200 cm^2 and weighs 17.4 grams. This cell is made with an 0.002 inch thick molybdenum substrate. A different and thinner material will reduce the cell weight to 15.9 grams in the future. Today's cadmium sulfide space solar cells are sandwiched between layers of Mylar$^{(R)}$ -

3-130

Nylon[R] combination. This material deteriorates under space ultraviolet such that the solar cell power output is down by 10 percent at the end of 1000 hours. A new plastic, which is in the development state, does not deteriorate under ultraviolet but has a significantly reduced initial transmission. A material which has high stable transmission is necessary before cadmium sulfide can be used in space. The use of this type material was assumed in the reference concept designs.

Absorptivity and emissivity of the front side of a cadmium sulfide cell is not known. The coating used on the back of solar cells has a measured emissivity of 0.88. In the reference concept designs the values used in the example of silicon solar cells was used.

The following are additional assumptions used in determining the two conceptual reference designs using cadmium sulfide solar cells.

1. Four 6 in. x 6 in. solar cells can be mounted in each square foot of array. The total active area per square foot is 800 cm^2 or 124 square inches.

2. The noon cell operating temperature is 128°C when the shield is used and 138°C when the shield is not used (BETA = 0.9 in Figure 3-40).

Table 3-6 gives a range of power output and cell weight (on a square foot basis) for assumptions which require definite technological advances. It can be seen that improvements in cadmium sulfide cells can make them competitive with todays silicon cells.

An array of cadmium sulfide solar cells may not need the same type panel which was used for silicon cells. This is because the cadmium sulfide cell has over 100 times the area of 1 cm by 2 cm silicon cells and has one half the unit area weight. A possible way to make a cadmium sulfide solar cell array is to join the edges of the encapsulating plastic of one cell to that of adjoining cells. This may then be mounted in a frame with additional support from cross members. Because of the reduced weight of cells and

lack of a panel as such, the weight per square foot of the cadmium sulfide solar cell array will be less than that for silicon.

TABLE 3-6

	Todays Array	Future Array
Solar Cell Efficiency @ 30°C	3%	5%
Solar Cell Efficiency @ 128°C	1.39%	3.39%
Solar Cell Efficiency @ 138°C	1.19%	3.19%
Spectrum Shift Factor	1.4	1.8
Power Output Watts/Sq. Ft. Area @ 128°C	1.55	4.88
Power Output Watts/Sq. Ft. Area @ 138°C	1.33	4.59
Cell Weight lb./sq. ft.	0.153	0.140
Cell Dimensions (nominal) - inches	3 x 3	6 x 6
Number of Cells/Sq. Ft.	16	4

SOLAR CELL ARRAY CONSIDERATIONS

Solar Cell Array Degradation

The information given thus far does not include any degradation. Some radiation in damaging amounts may be present. The effect of radiation may be taken into account by designing initially excess capacity into the system, designing in protection from radiation, or both. It should be noted that solar originated radiation will only effect the array during the sunlight portion of the mission (one half time) as the moon will shield the array during the other half time. At this time any effect of degradation may be taken into account by multiplying the watts per pound information in Tables 3-5 and 3-6 by a factor less than one and probably not less than 0.9 for a 3 year mission.

Cadmium sulfide solar cells appear to have a resistance to radiation which is superior to silicon cells. Figure 3-44 shows a comparison between silicon and cadmium sulfide solar cells bombarded with electrons. The effect is similar for proton bombardment.
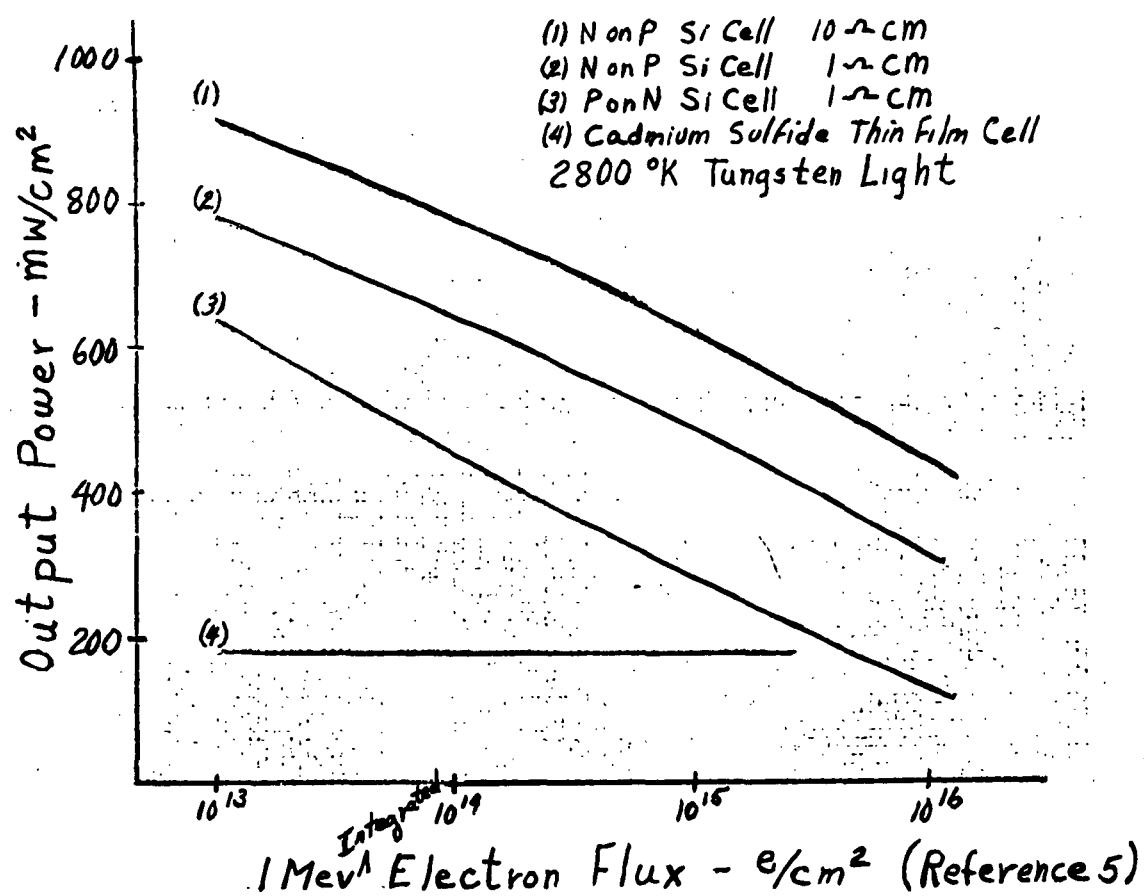
3-132

Figure 3-44. Solar Cell Degradation Characteristics

Panel Orientation

All power and efficiency information in this report is based on orientation where the sun line is normal to the plane of the array. Any deviation from this condition will cause a decrease in maximum power available. This decrease is approximately a function of the cosine of the referenced angle in the case of a flat panel. A ten degree error in this angle would cause the maximum power to decrease approximately 2.5%. The power decrease can be more severe under certain conditions when concentrators or reflectors are used.

The cosine effect just described eliminates the use of a flat panel mounted on the moon's surface. Panel orientation is necessary for a solar cell system to provide the maximum power with minimum fluctuations through the lunar day.

System Volume

The given reference concept designs have not included information on packaging, assembly tools, system packaged volume, etc. Packaged volume must include fixtures, boxes, etc., which are associated with packaging the array. In some instances the packaging may be a part of the vehicle. Packaged volume is also a function of the packing density which is related to the amount of work necessary for array erection.

A 1 cm by 2 cm silicon solar cell array will have a higher volume per unit panel area than will the tin film cells array if the technique in the reference conceptual designs are used. The packaged volume in general, will follow the array weight.

Cost

The cost of solar cells will be the main cost in assembling a solar cell array. The cost of solar cells and glass cover slips in the quantities necessary for kilowatt power supplies is estimated to be in the order of $5 per 1 cm by 2 cm size, or $2,100 per square foot. Cadmium sulfide solar cells

are presently available in 3 in. x 3 in. size for $30. If the improved cells anticipated for the future cost this much (and this is doubtful) then the square foot cost would be $480.

Reliability

A photovoltaic lunar power supply will use a large number of solar cells to meet the predicted power requirements. The number of solar cells connected in series and the number of solar cells connected in parallel are determined by the desired output voltage and output power respectively. It is common practice to place networks of the series-parallel arrangements on individual modules. Several of these modules are then placed in parallel through blocking diodes to a common bus. The parallel combination of modules make up the solar array. A system built around this design philosophy can tolerate several faults on the solar array and continue to operate at reduced capacity. The question concerning the desirability of using one large array or several smaller arrays can only be answered when specific load requirements and physical characteristics of the application become defined.

The output power of a photovoltaic system is affected by mission time and environmental conditions. The effect of solar cell degradation can be compensated for by designing added capacity into the system initially. The effect of meteoroids and possible lunar dust must be given consideration in preparing the surface of the array. The availability of humans on the moon will, of course, increase the reliability of all systems.

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INSTALLED CAPACITY REQUIREMENTS FOR A SOLAR ELECTRIC-POWER SOURCE

If a solar electric-power source is required to supply a relatively constant load during only daylight, the problem of installed capacity is straightforward. Where the load is highly variable or intermittant, however, it appears that a system employing energy storage may prove to be more economical from a weight standpoint. In addition, storage allows night-time operation.

Using the operation factors and load factors specified for the power demand models of (1) the roving vehicle, (2) the shelter, and (3) "miscellaneous", calculations have been made involving capacity of solar sources having storage capability. Figures 3-45 and 3-46 show solar-source capacity per unit
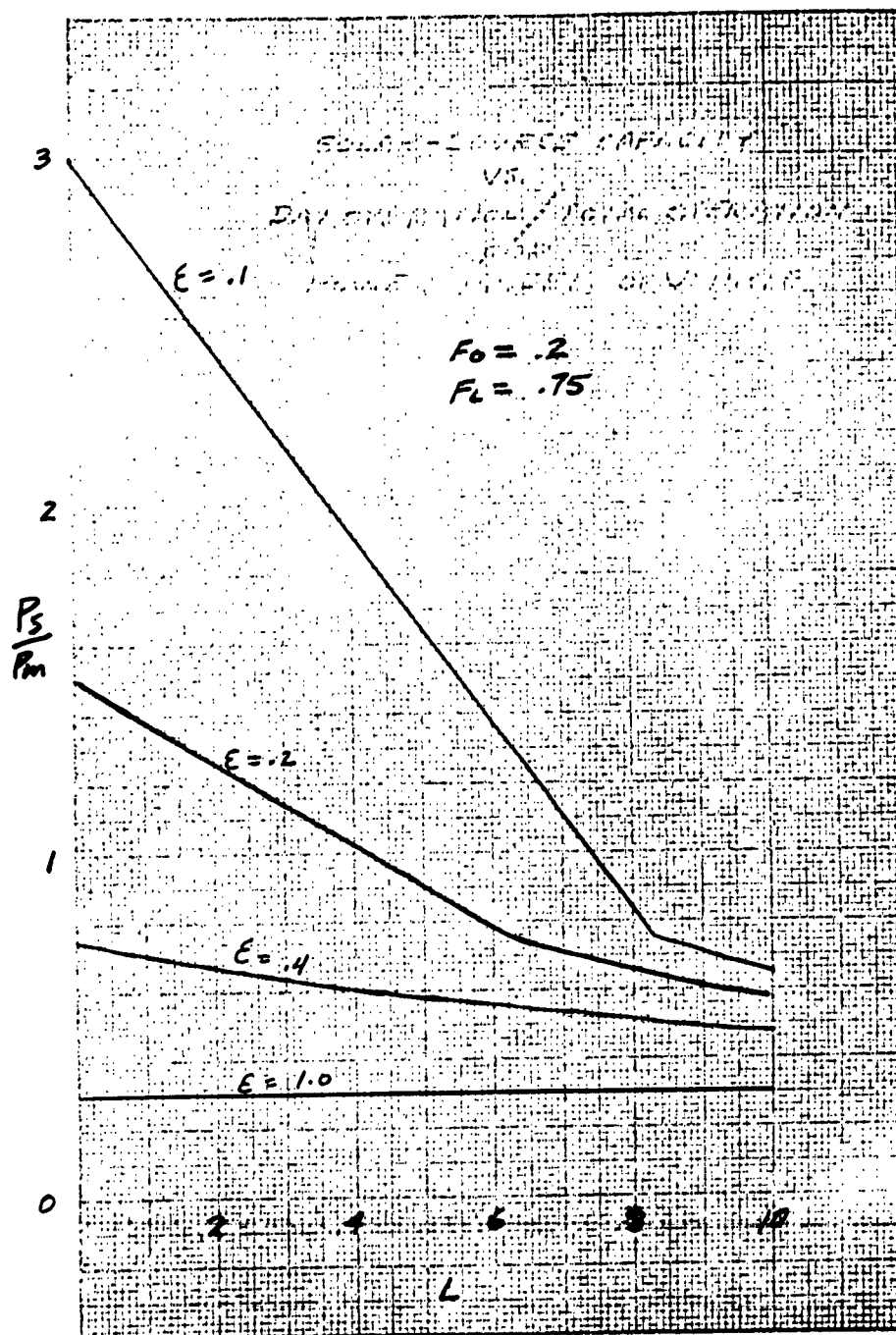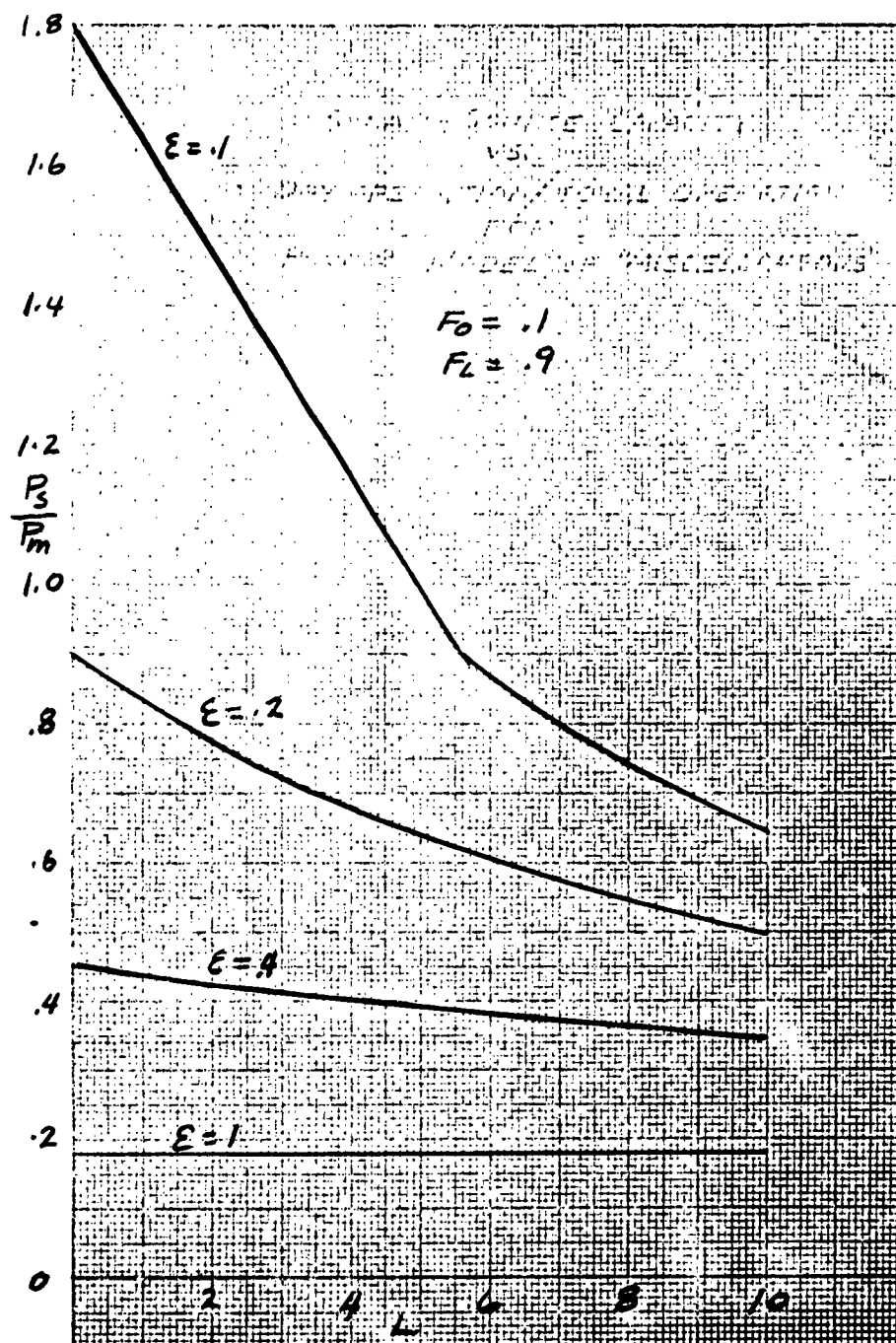
Figure 3-45.   Solar Source Capacity

Figure 3-46.   Solar Source Capacity

of maximum load as a function of that portion of operation occurring in day-light, with assumed storage efficiencies. As shown in these curves, storage efficiency and amount of night operation are both major factors in the capacity requirement of the solar source.

If all operation is confined to daylight such that $L = 1$, considerable reduction in solar-source capacity may be realized in a system having storage capability. In the case of the "miscellaneous" model, Figure 3-46, the solar-source capacity needs to be only 65 percent that required without storage, with only 10 percent storage efficiency. With 20 percent storage efficiency only 50 percent solar-source capacity is required.

As L is decreased (night operation increased) more capacity is, of course, required. The curves do not indicate the amount of storage capacity required, as this is a function of the load profile. The greatest storage demand may occur during night operation but not necessarily.

The shelter model with an operation factor of 0.99 is assumed to use equal amounts of energy both day and night, and thus the value of L is fixed at 0.5. Figure 3-47 shows the solar-source capacity per unit full load demand as a function of storage efficiency.

The curves of Figures 3-45, 3-46, and 3-47 are based on equations developed from the following assumptions and definitions:

P_S = Average power output capacity of the solar source during daylight. Total capacity is assumed to go to the load or to storage.

P_L = Average power to loads when loads are on.

P_M = Maximum load power required.

F_O = Use factor. Fraction of time that loads are connected to the system.

F_L = Load factor, P_L/P_M.

L = Ratio of daylight operating time to total operating time.

E = Average storage efficiency. Ratio of energy out of storage to energy supplied the storage device.

T = Operating period, a lunar day and night.

Energy required from the solar source for use during the lunar night is

$$(P_L/E) \, (1-L) \text{FoT}$$

Energy required from the solar source during the lunar day depends on the magnitude of the load relative to the solar-source capacity. If P_L P_S the load demands full capacity from the source plus power from storage. For this case, energy from the source is

$$\left(P_S + \frac{P_L - P_S}{E}\right) \text{FoLT}$$

For the case where P_L P_S the energy required is

$$P_L \text{FoLT}$$

The energy from the source which is in operation half the period is $P_S T/2$. For P_L P_S

$$P_S T/2 = \text{FoT} \left[\frac{P_L}{E} (1-L) + L \frac{(P_S + P_L - P_S)}{2} \right]$$

and

$$\frac{P_S}{P_M} = \frac{2 \text{FoF}_L}{E + 2\text{FoL}(1-E)} \tag{1}$$

Where P_L P_S

$$P_S T/2 = P_L \text{FoT}(L + 1/E - L/E)$$

and

$$P_S/P_M = 2\text{FoF}_L \, (1-L+EL)/E \tag{2}$$

The curves of Figure 3-45 and 3-46 make use of both equations (1) and (2), the transfer point being where $P_S/P_M = F_L$.

$$\frac{P_s}{P_m}$$

STORAGE EFICIENCY

$F_o = .99$
$F_o = .75$
$L = 1/2$
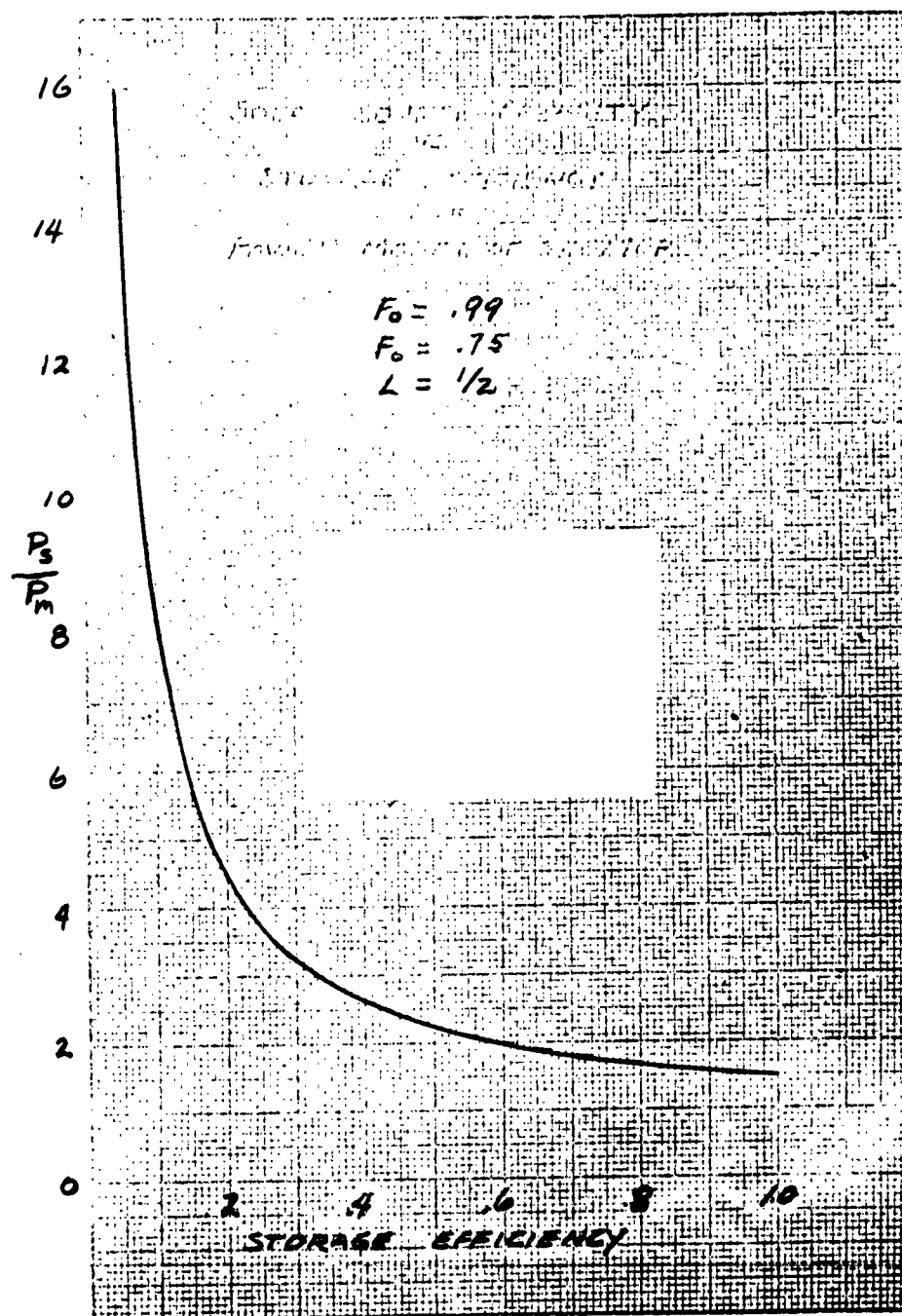
Figure 3-47. Solar Source Capacity

E. FLAT PLATE SOLAR THERMOELECTRIC GENERATORS

Flat-plate solar-thermoelectric generators usually consist of cylindrical thermoelectric pellets which are placed in the holes of aluminum honeycomb, and are sandwiched between two flat plates. The height of the honeycomb is approximately one half the pellet height and the honeycomb is attached to the bottom plate, the radiator, in order to give strength to the device. The top plate is the collector and has coating applied to it which has a high solar absorptivity to infrared emissivity ratio.

This absorptivity-emissivity ratio determines the equilibrium temperature of the collector. The power generated by the device is a function of the collector geometry and temperature, the thermoelectric material properties, the pellet geometry, and the radiator and heat-sink temperatures. The device can be designed for either high efficiency, watts/ft^2, or maximum specific power, watt/lb. When the collector coating has been specified, the maximum specific power is achieved by using very small diameter pellets, approximately 0.05 inches, with a pellet length of approximately twice the pellet diameter and a distance of sixteen diameters between pellets. The design which results in the maximum specific power has a very low efficiency. The efficiency may be increased for any given pellet diameter and spacing by simply increasing the pellet length. This, however, decreases the specific power.

Three small prototype flat-plate thermoelectric generators have been put in a satellite and tested. The efficiency of these devices was 1.5 percent and no data was given for the specific power. The power output degraded 50 percent after 40 orbits for one prototype and almost 100 percent for the other two. When problems such as very high thermal shunt losses from the collector to the radiator and inconsistencies in manufacturing of the thermoelectric pellets and joining techniques can be solved, it has been predicted that 10 watts/pound can be achieved from these devices. Other predictions

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of 20 watts/pound and higher have been made, but these predictions are overly optimistic.

All of these results are for flat-plate solar-thermoelectric generators which are in orbit or radiate the excess heat to deep space. If a device of this type were to be used for lunar-base power it would have to reject the excess heat directly to the surface of the moon or reject it to a fluid which could give the heat to a radiator which in turn could reject the heat to space. If the device rejected the heat directly to the surface of the moon via radiation, the temperature of the radiator would be much higher than if the device rejected the heat to deep space.

Since the collector temperature is not significantly different for either case, the temperature gradient across the thermoelectric material must be greatly reduced. This results in reducing the power output for the case where heat is rejected directly to the moon's surface. If a fluid and radiator were used to reject the excess heat, extra equipment and power must obviously be supplied. In both of these cases the efficiency and specific power would be less than if the device rejected the heat directly to deep space.

The optimistic prediction of 10 watts/pound which does not include power-conditioning equipment or supporting structures, and an efficiency of 1.5 percent could never be achieved in the necessary time period for lunar-base operation. For these reasons it has been concluded that flat-plate solar thermoelectric generators would not be feasible for supplying power for a lunar base.

SOLAR THERMOELECTRIC GENERATOR

In 1962, work was initiated by Westinghouse Aerospace Electrical Division under Air Force Contract AF 33(657)-8089 to accomplish an analytical and applied research program on a solar thermoelectric generator. The objective of this program was to demonstrate that a thermoelectric converter utilizing thermal energy storage and coupled with a solar collector is a

feasible system for generating electric power continuously during a 90 minute, earth orbit. Prior to this, government and industry sponsored research (References 1, 2 and 3) had been directed toward the development of solar thermoelectric generators which did not utilize thermal energy storage. Results from these programs indicate that thermal cycling which would be experienced during earth or similar orbits would cause rapid deterioration of the generator.

The experimental results from the present Westinghouse program demonstrates that a solar thermoelectric generator which utilizes thermal energy storage is a feasible space power system. Twenty-eight hours of operation were accumulated under simulated space environmental conditions. The model was subjected to three start-ups and shut-downs. Twelve simulated orbits were accumulated with the last nine imposed consecutively. The generator was operated for 22 hours in natural sunlight. The model was then placed on life test using electric heaters. After accumulating 946 hours of operation and 514 cycles, a temperature controller failed which allowed the generator to overheat. Data recorded prior to the failure showed the unit to have deteriorated approximately 15 percent from its original performance level. The generator experienced 8 complete start-ups without any adverse effects. Separate thermoelectric modules have successfully completed 7,000 hours of operation.

A solar thermoelectric generator operating on a lunar base must be capable of performing at least 12 start-ups per year. In view of performance demonstrated by the model previously discussed, it is logical to conclude that this requirement can be satisfied.

COMPONENT DESIGN

There are several collector concepts under development at the present time. The status of solar energy collector technology is summarized for the most part in Reference 4. This paper was presented in September 1962. Since

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this date, two additional concepts have been developed to a state where data on their capabilities exists. A light weight cone-and-column concept was presented by T. J. McCusher at the summer meeting of the AIAA (Reference 5). A 4 foot diameter unit is presently being constructed for NASA. It is felt that this approach will be most advantageous for collectors in the 60 to 100 ft diameter range. The operation of this concept is based upon extremely small gravity forces and it is doubtful that it would perform satisfactorily on the moon.

The Tapco Division of Thompson-Ramo-Wooldridge Inc., has constructed a new concept for NASA. The collector is composed of a single membrane shell (eight sectors joined at radial seams) of 0.016 inch thick aluminum alloy. Aluminum stiffener rings are joined to the back surface near the outside diameter. The collector is 5 ft in diameter and has a rim angle of 60 degrees. It was designed for 100 g inertia load in either direction along the optic axis. The weight of the unit is 0.57 lb per square foot of projected area. This weight includes the shell, rings, and brackets. Test results show that the cold calorimetric efficiency of the collector is equal to the solar reflectivity of the surface (88 to 90 percent) at a concentration ratio of 1,200 (concentrator area/cavity aperture area).

Table 3-7 gives a summary of weight and efficiency for collector concepts which are considered to be satisfactory for a solar thermoelectric generator.

Figure 3-48 illustrates a concept of a generator which is adaptable to a lunar station. This concept utilizes a cylinder for the cavity. The cylindrical cavity offers considerable design freedom since the internal dimensions may be changed without reflecting or radiating a large amount of the concentrated energy. There are 13 individual power modules in the generator. Twelve are located in the cylindrical portion and one is located in the rear portion of the generator. The waste heat is radiated from the outer frame and the twelve radial fins.

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An analysis of this design has been made to determine its capabilities. Table 3-8 gives the efficiency and other parameters of interest for this concept when used in conjunction with the single membrane collector given in Table 3-7. The recorded values are for a generator which has an 8 inch outside diameter excluding the radial fins.

TABLE 3-7

Collector	CR*-Efficiency	Unit Weight lb/ft^2	Specific Power Thermal Watts/lb.
FRESNEL	71% (Max)	0.46	201
Ref. 6	185 - 69%		195
	300 - 67%		189
	567 - 60%		169
	1000 - 49.5%		140
	1450 - 44%		124
Single Membrane	90% (Max)	0.57	205
Ref. 7	1000 - 90%		205
	1400 - 89.5%		204
	1600 - 89%		203
	1800 - 88%		201
	3000 - 80%		182
Electroformed Nickel	92.9% (Max)	0.70	172
Ref. 8	900 - 89.9%		167
	1175 - 92.9%		172
	1600 - 91.0%		169
	2295 - 89.0%		165
	3600 - 83.6%		155

*CR = Projected area of collector area of aperture.
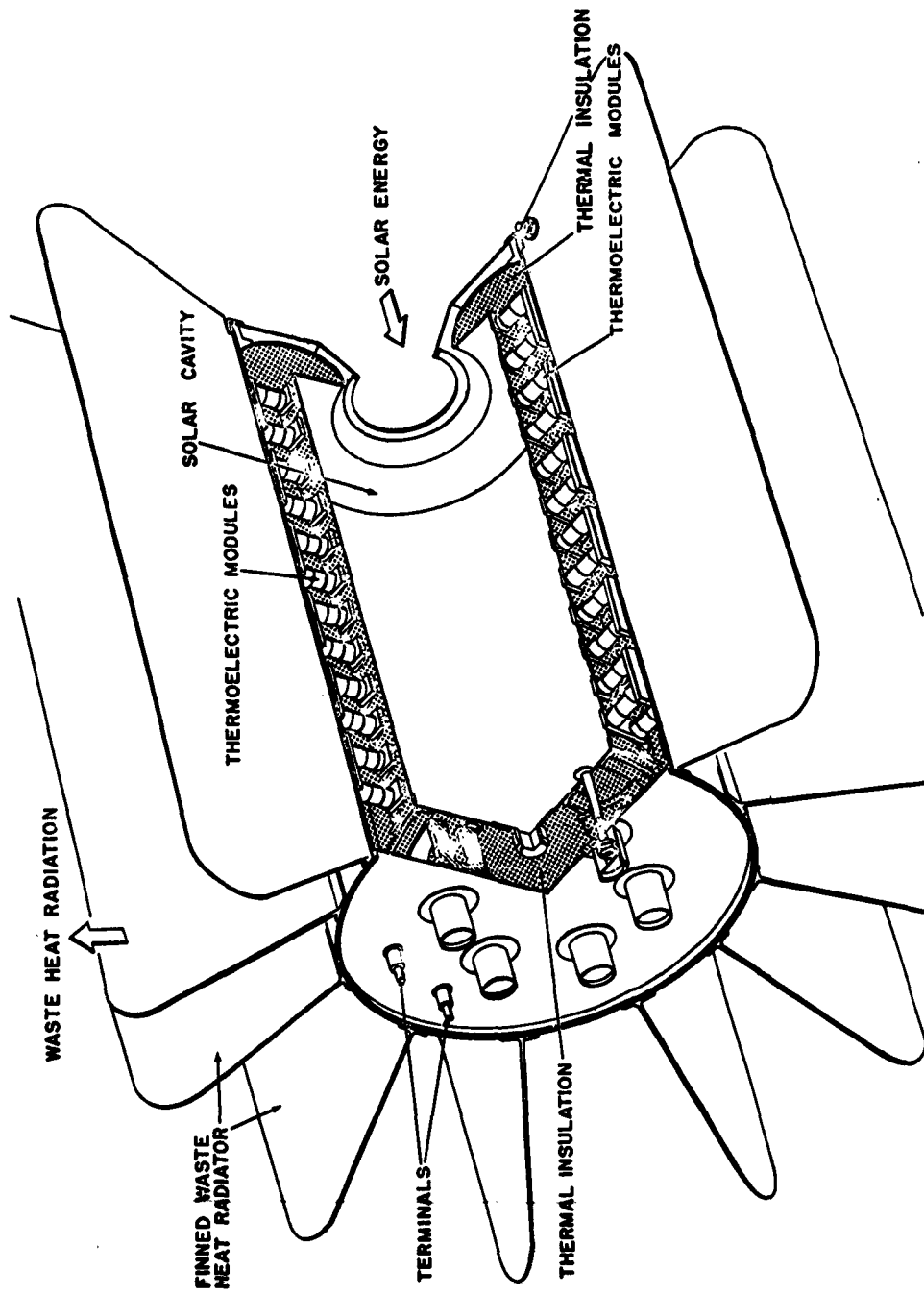
Figure 3-48. Concept for Solar T/E Generator Design

SOLAR ENERGY

THERMAL INSULATION

THERMOELECTRIC MODULES

SOLAR CAVITY

THERMOELECTRIC MODULES

WASTE HEAT RADIATION

FINNED WASTE
HEAT RADIATOR

TERMINALS

THERMAL INSULATION

TABLE 3-8

Power Output	57.5 watts	189 watts
Thermocouple Eff.	6.63%	6.63%
Gener. Eff. (including losses from cavity)	5.26%	5.46%
Collector Eff.	89%	89%
Over-all Eff.	4.68%	4.86%
Hot Junction Temp.	550°C	550°C
Cold Junction Temp.	210°C	210°C
Length of Generator	10.4 in.	30.3 in.
Length of Fins	5.04 in.	8.63 in.
Collector Dia.	3.47 ft.	6.16 ft.

The power to weight ratio of the collector and generator is given in Figure 3-49. It can be noted from this curve that specific power decreases when the power output of the device is increased. The decrease is due to an increase of the specific weight of the fins or radiator as shown by Figure 3-49. The fin weight increase is due to a decrease of the view-factor of the fins to the surroundings when the length to diameter ratio of the generator is increased.

Table 3-9 gives a weight summary for a unit which has an output of approximately 100 watts. These weights can be considered typical for a solar thermoelectric power system for a lunar power station.

EFFICIENCY, WEIGHT, AND POWER RANGE

The overall efficiency of a solar thermoelectric system can approach 4.7 percent. In order to obtain a reasonable power to weight (2.7 watts/pound), the output of an individual generator should be approximately 100 watts. This approach provides a greater reliability and requires less storage volume during launch. A system power output greater than 0.1 kw can be obtained by using individual units connected in parallel or in series.
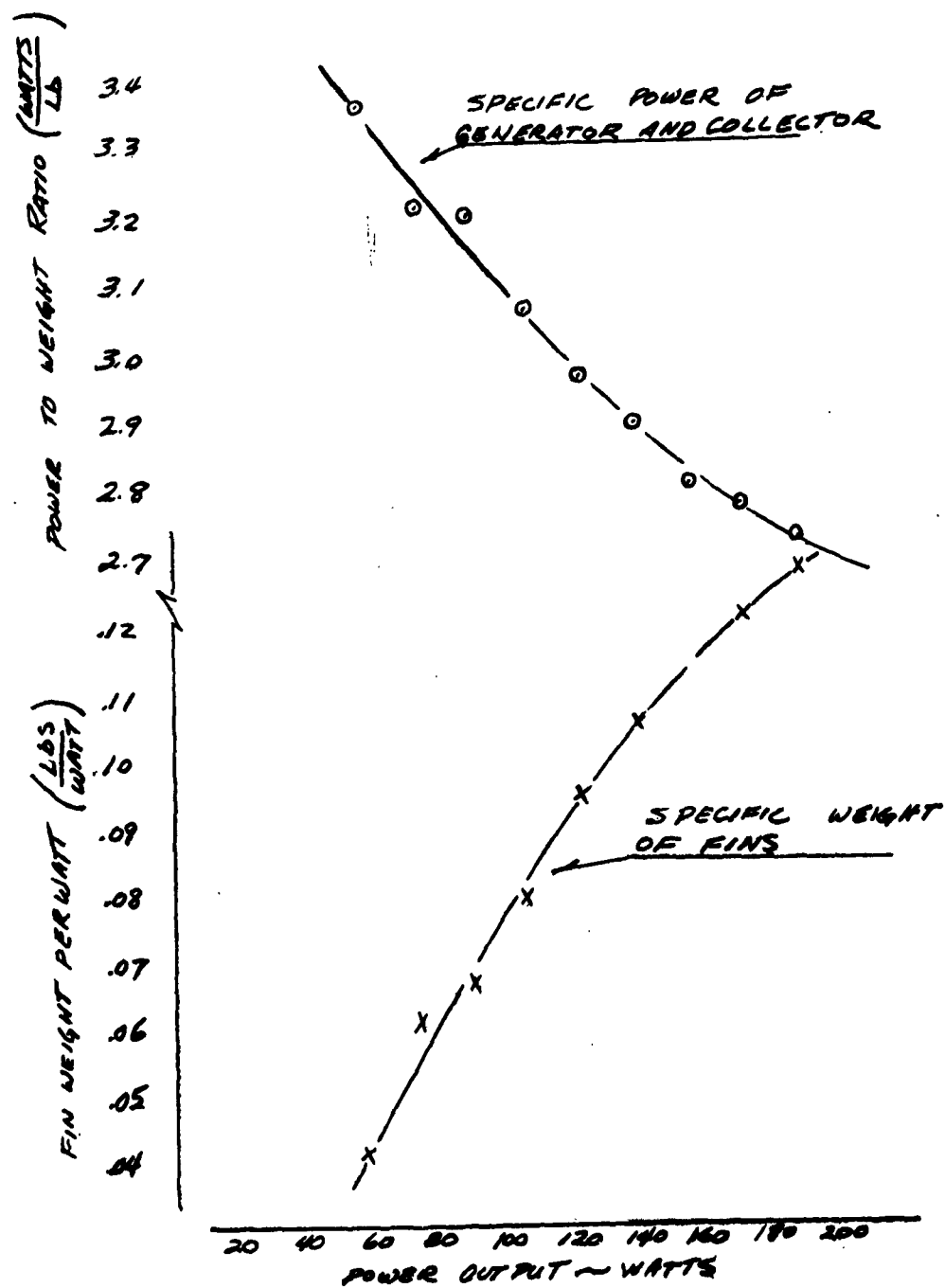
Figure 3-49

TABLE 3-9

Component	$\dfrac{\text{lb.}}{\text{Electrical Watts}}$	$\dfrac{\text{Electrical Watts}}{\text{lb.}}$
Radiator Fins	.0794	12.59
Radiator Cylinder	.0136	73.55
Cavity	.0237	42.78
Thermopile	.0863	11.59
Insulation	.0115	98.49
Hardware in Generator	.0190	52.77
Collector	.0760	13.16
Collector and Generator	.309	3.23
Mounting Structure and Orientation System	.060	16.66
Total System	.369	2.7

PACKAGED VOLUME

The volume or envelope of a generator when mounted on a one piece collector is much greater than the volume of the same components when they are not assembled. It therefore appears feasible to launch these components unassembled. Several generators can be stored in a small volume. Likewise, a large number of collectors can be stored in another compartment of the launch vehicle. This approach offers considerable freedom for design of the support structure within the vehicle. It is reasonable to assume that this design freedom is required since considerable inertia loads will be imposed on the system during launch and landing. Table 3-10 gives the estimated volume requirements of components for a 1 kw system.

TABLE 3-10

Component	Vol (ft^3)
Collectors	86
Generators	41
Mounting Structure and Orientation System	10
Total	137

RELIABILITY

The reliability of a thermoelectric conversion system must be considered in light of two types of failures. The first type of failure is one that results in zero power output from a part or the entire device. A failure of a component within the thermopile which would cause an open circuit can be considered an example of this kind of failure. If the failure occurred in a module that could be removed or isolated from the system, it would not render the entire conversion system useless which would constitute a partial failure. The second type of failure is that which results in a degradation of the power output of the device until it is below the specified requirements.

The probability of failure of the first type can be controlled by good design technique. Once a design has been developed and acceptance tested, failure of this kind can be predicted with reasonable accuracy. The penalty of this type of failure can be decreased by designing the system so that faulty components can be electrically isolated from the rest of the system. Full mission requirements can be maintained by providing an initial over-design of the system.

The majority of the failures that occur in thermoelectric conversion devices result from failures of the second type. Degradation of output power is due primarily to attack by oxygen. The problem of oxidation can be overcome by utilizing a sealed thermopile construction that incorporates an inert or

reducing atmosphere. This inert atmosphere eliminates oxidation during storage and operation in the earth's atmosphere prior to launch. Stresses induced by thermal cycling and the maximum operating temperature of the thermoelectric material also effect the life of the device. The proper selection of couple length, operating temperatures, and start-up procedure will increase system reliability.

Existing designs have demonstrated over eight thousand hours of operation with approximately 15 percent degradation of initial output. Current data indicates that the trend of degradation might level off at 20 to 25 percent for a three year mission. The reliability of the system is increased by providing an initial over-design to compensate for types of failure discussed above. Overall mission requirement can best be satisfied by using several generators with each one coupled to a collector. Individual generators should also have modules which can be electrically removed from the thermopile. The initial total power output of the modules must exceed the output requirements at the end of a mission by the amounts given below.

Operating Time In Years	Over-Design Requirement
0.5	13.2%
1.0	18.0%
1.5	20.7%
2.0	22.6%
2.5	24.0%
3.0	25.0%

OPERATING PROCEDURE

It does not appear to be practical to build a complete solar thermoelectric system that would be self erected. A practical system would require the following field operation before the system could function.

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1. Erection of the support structure and orientation mechanism.
2. Mounting of collectors and generators on the support structure.
3. The array of collectors and generators must be manually orientated so that rotation about only one axis is necessary for following the sun.

Practical aspects from the standpoint of system reliability will require some maintenance of the system. Provisions to permit replacement or repair of a faulty component will be required.

COST CONSIDERATIONS

It is not possible to estimate the development cost of this type of system since the specifications have not been defined. If one considers only the manufacturing and material cost of a system after it has been developed, one can estimate a cost per unit of electric power. This cost has been estimated to be $100 per watt.

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IV. ENERGY SOURCES

A. CHEMICAL PROPELLANTS

BIPROPELLANTS

In the report for the period ending September 30, 1963, the fuels and oxidizers considered applicable to the lunar engine-fuel system study were listed. Also in that report a comparison was made between oxygen base and halogen base oxidizers, and it was concluded that the halogens should be eliminated from further consideration.

During the month of October, combinations of fuels and oxidizers were selected for evaluation in application to various types of thermal engines. Table 4-1 lists the combinations of fuels and oxidizers selected and gives the most significant properties of the products of combustion. These combinations were selected on the basis of storage compatibility and product composition.

An IBM 7090 computer program was used to calculate the thermochemical properties of the products of combustion of all of the fuel-oxidizer combinations at 77°F inlet and 500 psia pressure except for 6 and 7. These could not be calculated because the program is not set up to handle Cl and B. For the seven combinations, which have the flame temperature given, working plots of molecular weight, ratio of specific heats, and flame temperature were made as a function of the oxidant-fuel ratio. Similar plots for the other fuel-oxidizer combinations will be made during the month of November.

For an internal combustion engine, where the combustion products are expanded in a cylinder or through a turbine, the ideal heat to work in the expansion is approximated by the following expression.

TABLE 4-1

		1	2	3	4	5	6	7	8	9	10	11	12	13	14
	Reaction	$H_2 + O_2$	$H_2 + H_2O_2$	$NH_3 + H_2O_2$	$N_2H_4 + H_2O_2$	$N_2H_4 + N_2O_4$	$N_2H_4 + NO_2ClO_4$	$B_5H_9 + N_2O_4$	$C_8H_{18} + H_2O_2$	$C_6H_5NH_2 + H_2O_2$	$C_2H_5OH + H_2O_2$	$NH_3 + O_2$	UDMH + N_2O_4	$CH_4 + O_2$	NH_3 + 50% H_2O_2
	Fuel	H_2	H_2	NH_3	N_2H_4	N_2H_4	N_2H_4	B_5H_9	C_8H_{18}	$C_6H_5NH_2$	C_2H_5OH	NH_3	UDMH	CH_4	NH_3
	Molecular Weight	2.016	2.016	17.03	32.05	32.05	32.05	63.17	114.2	93.06	46.06	17.03	60.1	16.04	17.03
ΔH_{f298}	Heat of Form - BTU/# Mole	0	0	-28,600	21,700	21,700	21,700	13,900	-107,300	-4,160	-119,200	-28,600	-35,200	-32,224	-28,600
	Oxidizer	O_2	H_2O_2	H_2O_2	H_2O_2	N_2O_4	NO_2ClO_4	N_2O_4	H_2O_2	H_2O_2	H_2O_2	O_2	N_2O_4	O_2	50% H_2O_2
	Molecular Weight	32.00	34.02	34.02	34.02	92.02	144.5	32.05	34.02	34.02	34.02	32.00	92.02	32.00	23.55
ΔH_{f298}	Heat of Form - BTU/# Mole	0 Liq.	-80,600 Liq.	-80,600 Liq.	-80,600 Liq.	-12,300 Liq.	15,700 Solid	21,700 Liq.	-80,600 Liq.	-80,600 Liq.	-80,600 Liq.	0	-12,300 Liq.	0	-307,750 Liq.
	Stoichiometric Ratio -O/F Molar	1:2	1:1	3:2	2:1	1:2	4:13	5:2	25:1	15.5:1	6:1	3:4	2:1	2:1	4.33:1
	Stoichiometric Ratio -O/F Weight	8:1	17:1	3:1	2 1/8:1	1.44:1	1.39:1	1.27:1	7.45:1	5.66:1	4.43:1	1.41:1	3.06:1	4:1	5.98:1
	Heat of Reaction - BTU	-208,130	-127,530	-325,390	-276,760	-467,360	-3,000,712	-1,221,300	-2,770,800	-3,479,870	-872,505	-509,990	-465,180	-690,532	-223,000
	Heat of Reaction - BTU/#	-5,780	-3550	-2390	-2764	-2995	-3020	-4,260	-2875	-2805	-2640	-3105	-2850	-6310	-496
	Products n_1	H_2O	H_2O	H_2O	H_2O	H_2O	H_2O	H_2	H_2O	H_2O	H_2O	H_2O	H_2O	H_2O	H_2O
	n_2			N_2	N_2	N_2	HCl	BN	CO_2	CO_2	CO_2	N_2	CO_2	CO_2	N_2
	n_3						N_2			N_2			N_2		
Mp	Molecular Weight	18.0	18.0	19.4	30.0	22.2	23.2	9.9	NIL	NIL	NIL	20.5	NIL	26.6	NIL
	Flame Temp. - °F	5800	4780	4200	4750	5250						5000		5700	
γ_p	Ratio of Spec. Hts.	1.235	1.235	1.245	1.25	1.27	1.30=1600	1.29=1700				1.255		1.285	
	Corrosiveness	NIL	NIL	NIL	Some HNO₃ for φ > 1.0	NIL	Some HCl	NIL	NIL	NIL	NIL	NIL	NIL	NIL	NIL
	Toxicity	NIL	NIL	Some NO	Some NO	Some NO	Some NO₃Cl	NIL	NIL	NIL	NIL	Some NO	CO	CO	Some NO
	Recovery Tech.	Cond.	Cond.	Cond. & Absorp.	Cond. & Absorp.	Cond. & Absorp.	Complex	Complex	Complex	Complex	Complex	Cond. Absorp.	Complex		Cond. Absorp.
	Heat Reqd. - BTU/#	-1000	-1000	-1000/H₂O	-1000/H₂O	-1000/H₂O						-1000/H₂O			-1000/H₂O
	Regeneration Processes	Elect. Ox. of H₂O	Elect. Ox. of H₂O	Elect. Syn. of NH₃ Ox. of H₂O	Elect. Syn. of N₂H₂ Ox. of H₂O	Complex	Complex	Complex	Complex	Complex	Complex	Elect. Syn. of NH₃	Complex	Elect. Syn. of CH₄	Elect. Syn. of NH₃ Ox. of H₂O
	Energy Req. - kw-hr/#	4.0 - 4.4										2.1 - 2.4			
	Ratio of Heat of Reaction to Energy For Regeneration	.38 - .42										.38 - .43			
	Heat Rej. - BTU/#	9100 - 9600										4880 - 4350			

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$$\Delta h' = \frac{1544\ T_1}{778\ M_1} \left|\frac{\gamma_1}{\gamma_1 - 1}\right| \left[1 - \left|\frac{P_2}{P_1}\right| \frac{\gamma_1 - 1}{\gamma_1}\right]$$

where

$\Delta h'$ = isentropic heat to work - BTU/pound

T_1 = inlet temperature (^{O}R)

M_1 = inlet to molecular weight

γ_1 = inlet ratio of specific heats (BTU/pound/^{O}F)

P_1 = inlet pressure

P_2 = outlet pressure

If the work required to get the reactants up to P_1 is negligible, the isentropic specific propellant consumption of the engine is approximately equal to $3415/\Delta h'$ lb/kwhr. Most if not all of the internal combustion engines being considered for space or lunar applications are of this type. Therefore, a comparison of the different fuel-oxidant combinations on this basis is significant.

One of the prime considerations matching the fuel-oxidizer combination with an engine type is the limitation on maximum gas temperature. Piston type engines can handle gas temperatures in the order of 4500^OF, and turbines are able to withstand temperatures up to about 2000^OF. Inspection of the flame temperatures listed in Table 4-1 shows that stoichiometric proportions of most combinations give temperatures greater than 4500^OF.

There are four ways to moderate the gas temperature in a combustion process:

1. Use excess fuel
2. Use excess oxidant
3. Recycle products of combustion
4. Use a third gas as a dilutent

The third method would probably be advantageous only if the products could be condensed and pumped rather than compressed as a gas. The fourth method is unattractive because of the complexity of handling a third material.

During the first phase of lunar operations, when the products of combustion will not be recovered, temperature moderation would most logically be accomplished by using excess fuel or oxidant. The working plots of molecular weight and ratio of specific heats from the computer program were used on this basis to calculate the isentropic SPC for the seven combinations of fuels and oxidants. The reactant which had the lower molecular weight was used as a dilutent. The calculations were made for P_2/P_1 values ranging from 10^{-4} to 10^{-1}, and for inlet temperatures ranging from 1500 to 4500°F.

The results of these calculations are shown plotted in Figures 4-1 through 4-7. Inspection of those plots shows that H_2 - O_2 is best by a wide margin, H_2-H_2O_2 is next best, and the rest are about equal.

In these reaction calculations hydrogen, oxygen and methane, which probably will be stored at cryogenic temperatures, are assumed to be gaseous at 77°F. This is considered realistic because the difference between the storage state and the gaseous state at 77°F will be utilized as a heat sink.

MONOPROPELLANTS

Hydrazine, hydrogen peroxide, and ethylene oxide have been used successfully as monopropellant gas producers to rive turbine or piston expanders. Ethylene oxide has been found to give substantially poorer performance than hydrazine, so will not be considered here. The properties of hydrazine and hydrogen peroxide were given in Table 5 of the report for period ending September 30, 1963.

Over a wide range of concentration, hydrogen peroxide can be decomposed catalytically to give H_2O and O_2. The flame temperature varies linearly from 212°F for 65 percent H_2O_2 to 1825°F for 100 percent H_2O_2 if the initial condition is liquid at 68°F.
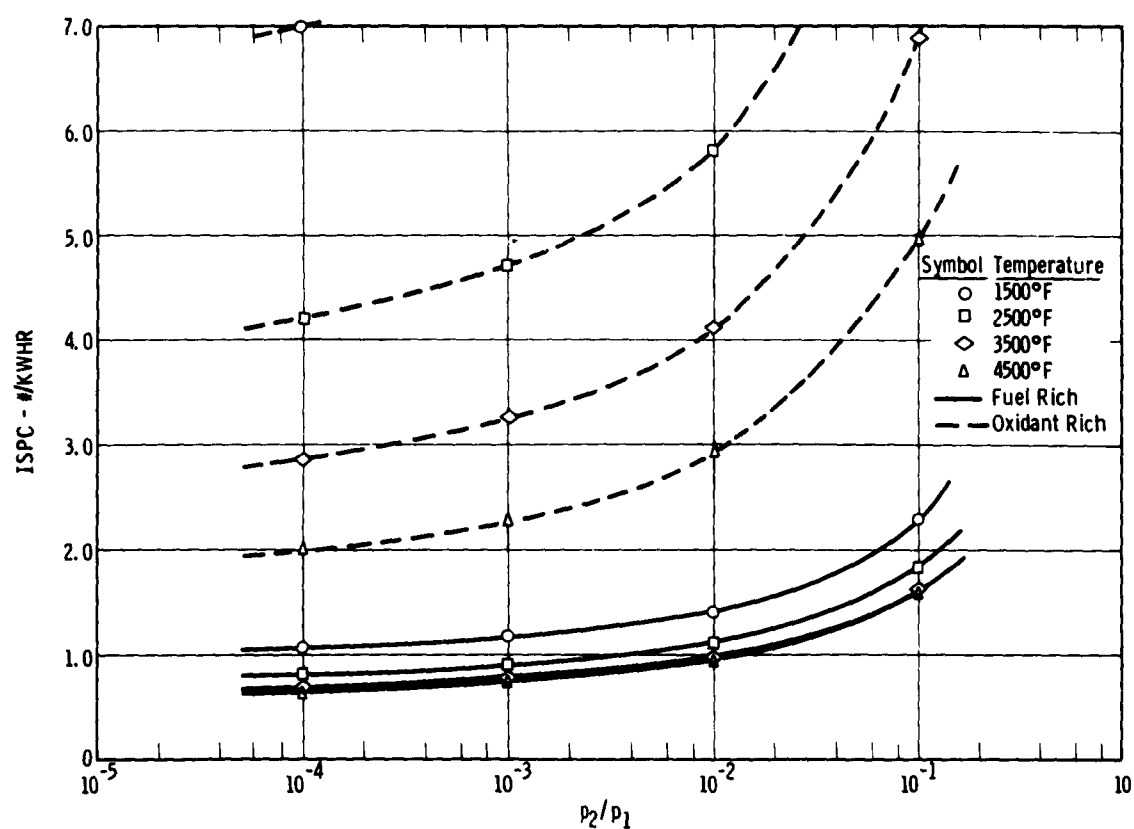
Figure 4-1. Isentropic SPC for Products of Combustion of Mixtures of H_2 - O_2 at $77^{\circ}F$
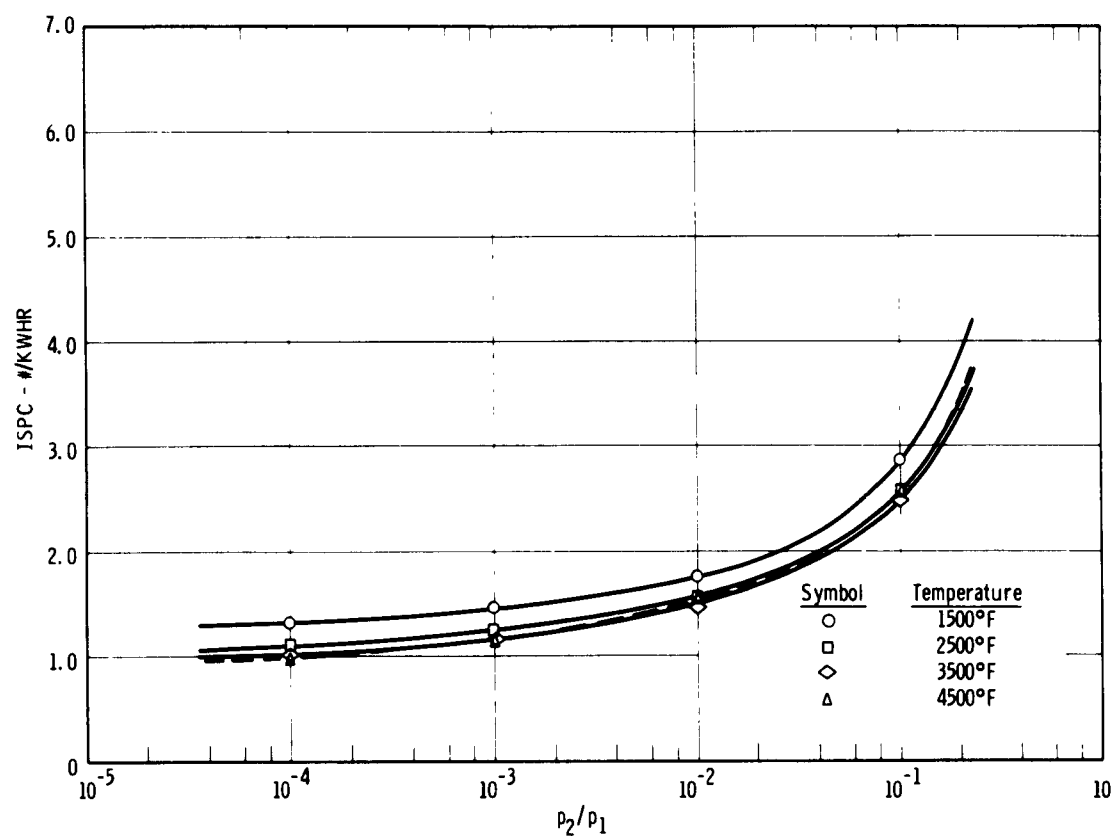
Figure 4-2. Isentropic SPC for Products of Combustion of Rich Mixtures
of H_2 - H_2O_2 (100%) at 77°F
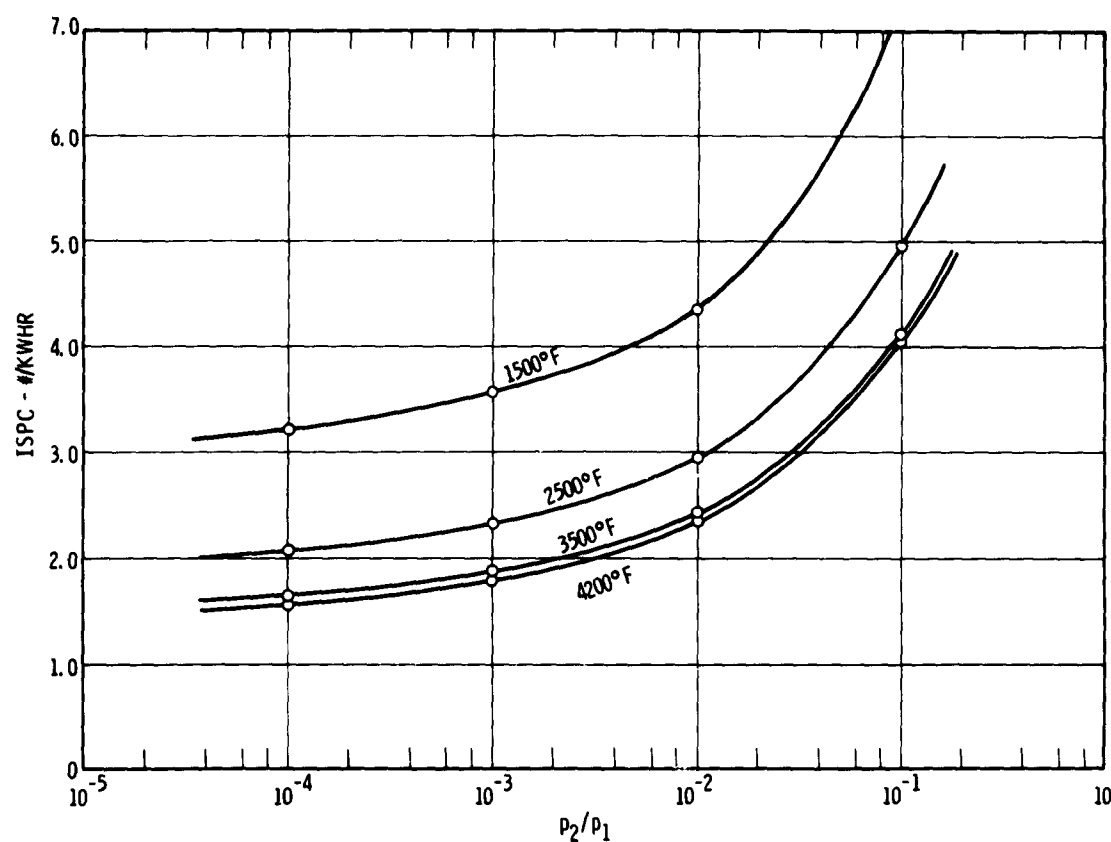
Isentropic SPC for products of combustion of fuel rich mixtures of 100% H_2O_2 & NH_3 at 77°F

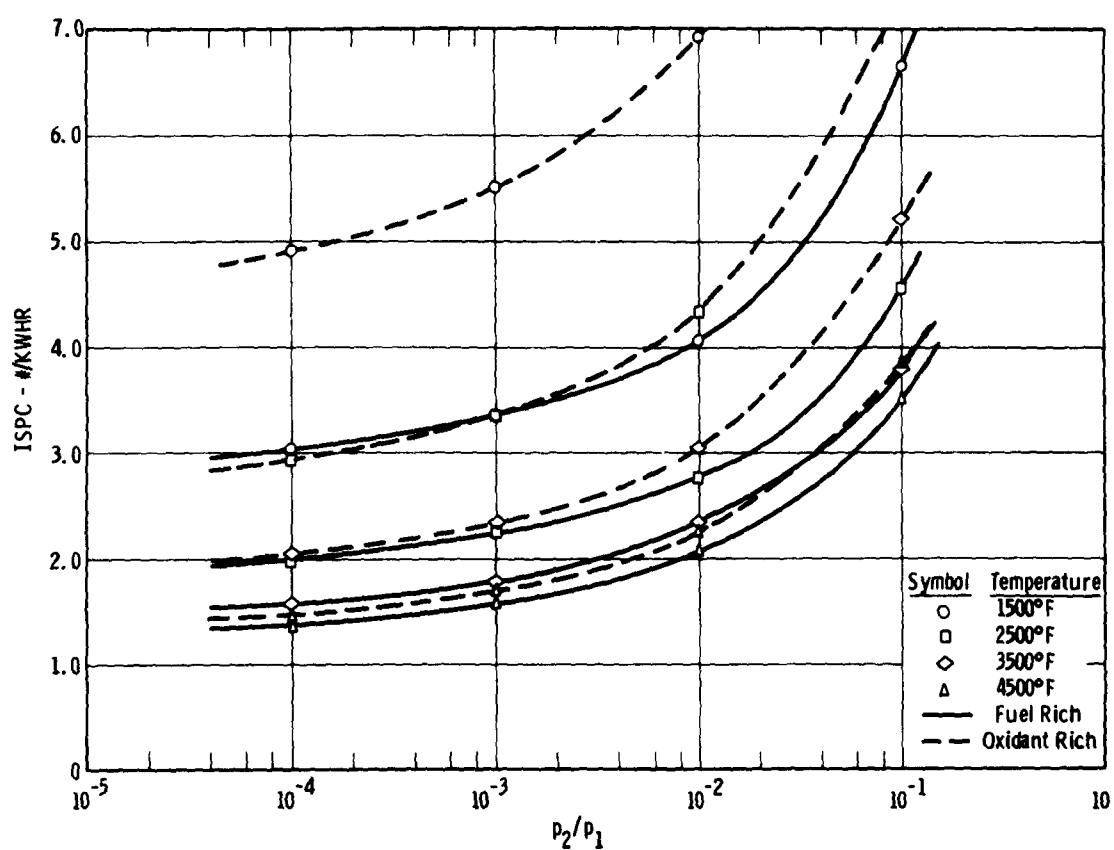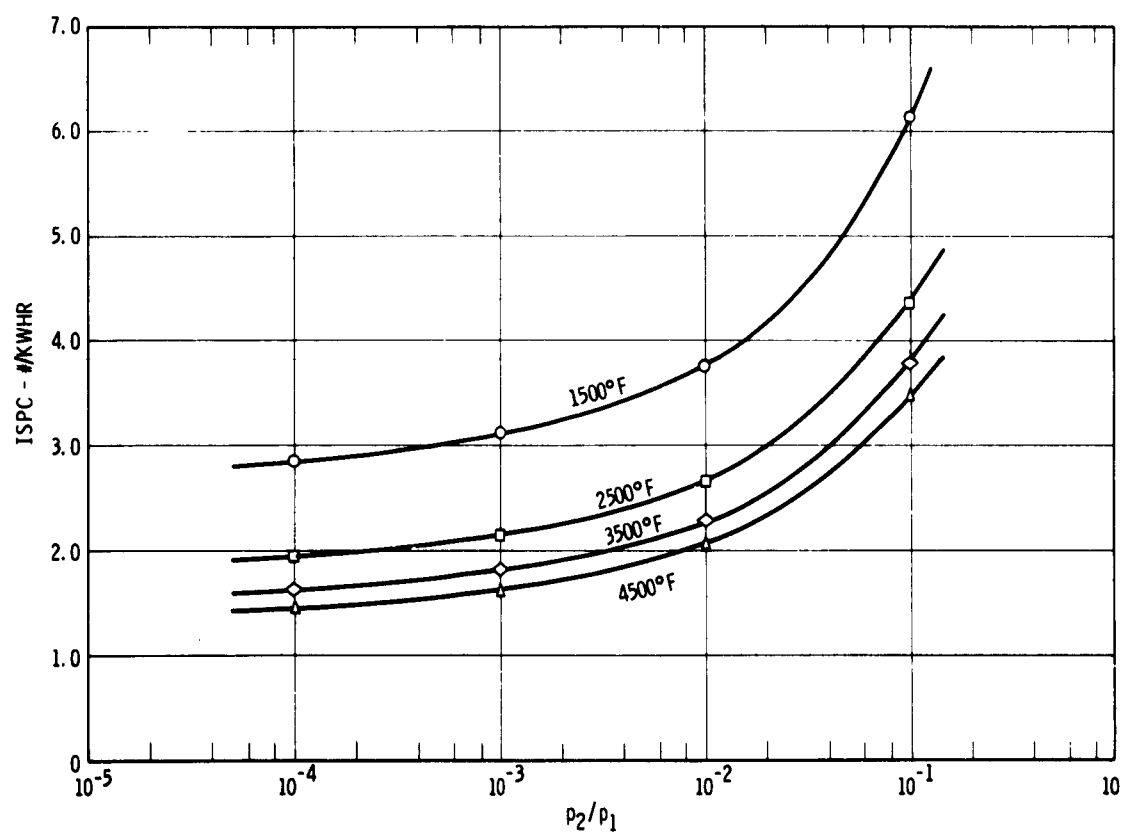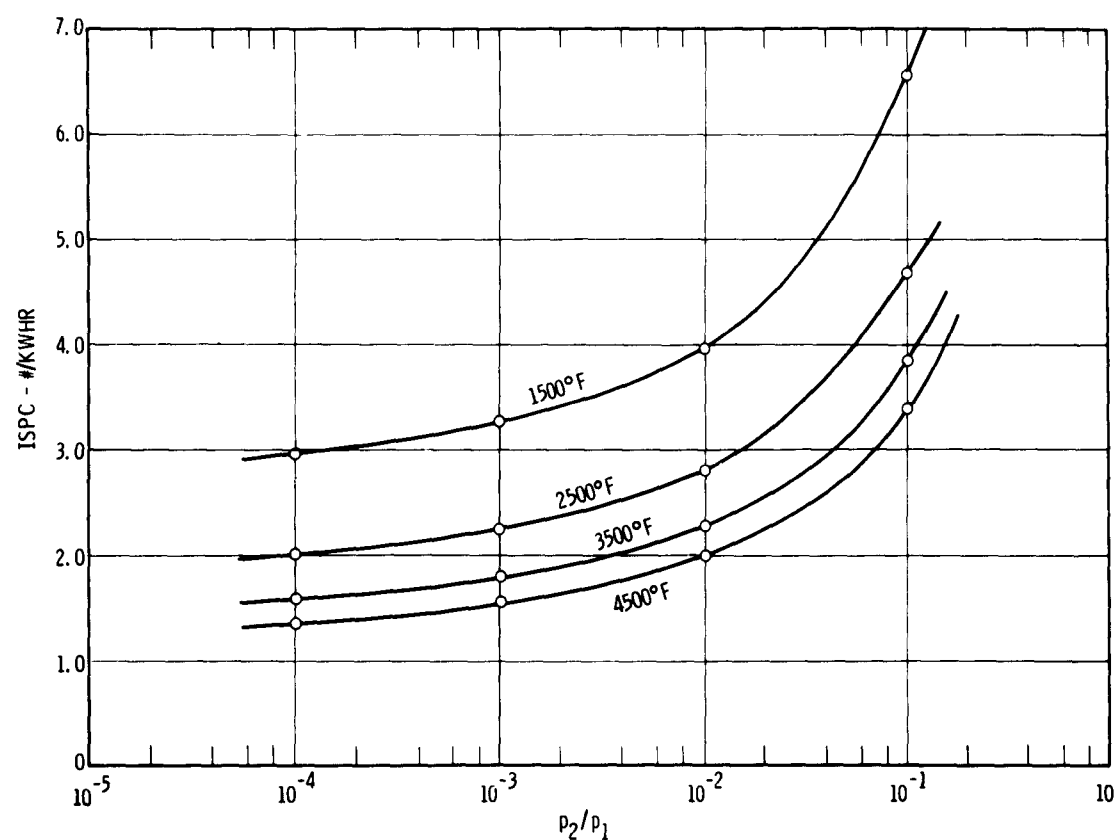Figure 4-3.   Isentropic SPC for Products of Combustion of Fuel Rich Mixtures of 100% H_2O_2 and NH_3 at   77°F

Figure 4-4. Isentropic SPC for Products of Combustion of Mixtures of
N_2H_4 - 100% H_2O_2 at 77°F

Figure 4-5.   Isentropic SPC for Products of Combustion of Fuel Rich
Mixtures of N_2H_4 - N_2O_4 at 77^OF

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Figure 4-6. Isentropic SPC for Products of Combustion of Fuel Rich
Mixtures of NH_3 and O_2 at $77^O F$
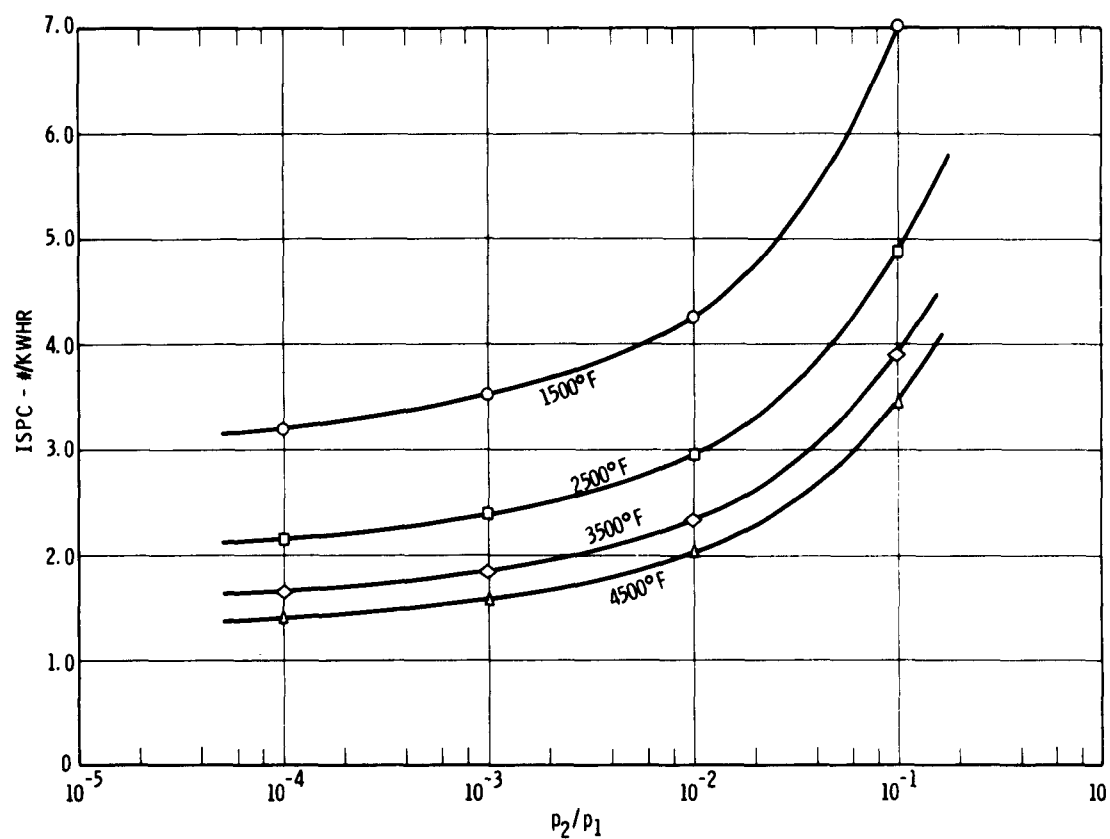
Figure 4-7. Isentropic SPC for Products of Combustion of Fuel Rich
Mixtures of CH_4 - O_2 at 77^OF

Hydrazine decomposes thermally into a gaseous mixture of ammonia, nitrogen and hydrogen with a flame temperature of 2050°F. This corresponds to a 25 percent dissociation of the ammonia into nitrogen and hydrogen.[1]

Table 4-2 gives thermochemical properties of decomposition products for 100% hydrogen peroxide and hydrazine, and isentropic SPC's for the expansion of these products through a turbine or piston expander. A comparison of these isentropic SPC's shows that hydrazine is clearly superior to hydrogen peroxide as a monopropellant gas producer.

RECOVERY OF PRODUCTS OF COMBUSTION

Among the combinations of fuels and oxidizers for which regeneration is considered feasible there are only four elements: N, O, C, and H. The products of combustion would thus contain mainly H_2O, CO_2, CO, and N_2. The recovery of water would most likely be done by condensation. The recovery of the other compounds is not so easy.

Following is a discussion of possible methods of recovering CO_2, CO, and N_2 in addition to H_2O.

ENGINE EXHAUST GASES RECOVERY

To be potentially free of the earth for supply of engine propellants, it is necessary to recover the engine exhaust gases and to reconstitute the propellants from them. It is therefore necessary to consider the problems and magnitude of the effort associated with this recovery or collection.

All processes pertinent to this effort may be divided into the three groups: adsorption processes, chemical processes, and physical processes. Since these are radically different, it is advisable to consider each separately.

[1]Zwick, Eugene B. "Chemical-Mechanical Space Power Systems." Progress in Astronautics and Rocketry - Vol. 4, Space Power Systems, p. 572.

TABLE 4-2

PROPERTIES OF DECOMPOSITION PRODUCTS OF MONOPROPELLANTS

	100% Hydrogen Peroxide	Hydrazine
Chemical Formula	H_2O_2	N_2H_4
Products		
η_1	H_2O	NH_3
η_2	O_2	N_2
η_3	-	H_2
Lower Heat of Decomposition of liquid @ 77°F BTU/lb	-690*	-1140**
Molecular Wt.	22.68*	14.2**
Flame Temp. -°F	1825*	2050***
Ratio of Spec. Hts.	1.248*	1.279**
Corrosiveness Toxicity	High on Steel NIL	NIL
ISPC - lb/kwhr		
P_2/P_1		
10-4	4.07	2.48
10-3	4.59	2.78
10-2	5.82	3.47
10-1	9.75	5.74

*Becco Div., FMC Corp.

**Calculated on basis of 25% decomposition of NH_3

***Zwick

The aim of any successful collection scheme is probably at least two-fold. First, it is required that the combustion products be collected and contained in as small a volume and with equipment having as low a complexity and weight as possible. Secondly, since collection connotes full reconstitution, the process should be such as to give up the combustion products without introducing too high an energy or equipment barrier. There are other requirements subsidiary to (and inherent in) the above two, but these will appear as the discussion proceeds.

Process Requirements

To be reasonably explicit in the recovery processes, it is desirable to make an assumption concerning the composition and quantity of exhaust gases. It will be assumed that the combustion products consist of N_2, H_2O, CO_2 and CO. Composition and quantity require certain assumed reactions between fuels and oxidizers. For this purpose, the fuels are NH_3, N_2H_4, CH_4, and aniline $(C_6H_5NH_2)$. The oxidizers are O_2 and H_2O_2.

The combustion reactions are

$$N_2H_4 + 2\ H_2O_2 = N_2 + 4\ H_2O \tag{1}$$
$$N_2H_4 + O_2 = N_2 + 2\ H_2O \tag{2}$$
$$2NH_3 + 3\text{-}1/2\ O_2 = N_2 + 3\ H_2O \tag{3}$$
$$2NH_3 + 3H_2O_2 = N_2 + 6\ H_2O \tag{4}$$
$$2C_6H_5NH_2 + 3\text{-}1/2\ O_2 = N_2 + 12CO_2 + 7\ H_2O \tag{5}$$
$$2C_6H_5NH_2 + 31\ H_2O_2 = N_2 + 12CO_2 + 38\ H_2O \tag{6}$$
$$CH_4 + 2O_2 = CO_2 + 2H_2O \tag{7}$$
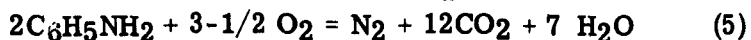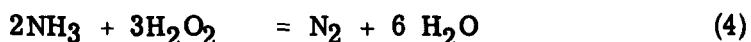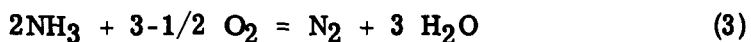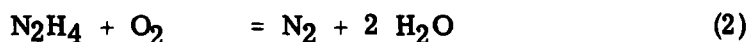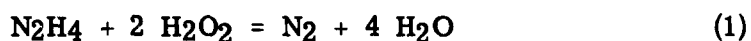
Heats of reaction for the seven reactions were obtained from tabulated heats of formation. These values were used to obtain the quantity of combustion products for the liberation of 1 kwhr of energy. This, of course, assumes 100 percent efficiency in the engine system. These data are shown in Table 4-3.

TABLE 4-3

QUANTITY OF COMBUSTION PRODUCTS

FOR 1 KWH AT 100% EFFICIENCY

Reaction	Volume of Products (Liters at S.T.P.)			Weight of Products (grams)		
	N_2	H_2O	CO_2	N_2	H_2O	CO_2
1	102	406		126	326	
2	140	278		174	212	
3	128	380		158	306	
4	84	508		106	408	
5	6	88	150	16	70	288
6	4	316	150	10	254	98
7		204	102		164	100

There is an error involved in the assumption of reactions 6, 7 and 8 where CO_2 is a product of combustion. We assume that the exhaust gas will contain CO. It is obvious that if conditions are not ideal, these reactions will produce CO as well as CO_2. This will, of course, result in a change in the data in Table 4-3 for these reactions.

To obtain the true state of affairs, assumptions would have to be made concerning the fuel-to-oxidizer ratio, burning conditions, engine expansion conditions, exhaust conditions, and engine efficiency. For this purpose a complete engineering study would be required. At the present stage of this work, such a study would hardly be reasonable for the purpose of this discussion. Here we are concerned not with the profound engineering picture, but with the overall broad recovery picture in order to obtain a fuel for the system.

Consequently, we shall make a crude assumption concerning reactions 6,7, and 8. We will retain the same amount of N_2 and H_2O. Since one volume of CO_2 converts

to one volume of CO, we shall assume that 30 percent of the CO_2 goes to CO. This gives the modified data of Table 4-4.

TABLE 4-4

QUANTITY OF COMBUSTION PRODUCTS

ASSUMING 30% OF CO_2 IN TABLE I CONVERTS TO CO

Reaction	Weight of Products (grams)			
	N_2	H_2O	CO_2	CO
5	16	70	160	43
6	10	254	78	13
7		164	70	19

Adsorption Processes

The kind of adsorption - whether van der Waal or chemisorption - is not of importance here, since we are concerned with the practical aspect of how much compound can be taken up by what volume of adsorbent in a reasonable time. For our purpose the adsorbent will be operating under a severe requirement.

In general, adsorption is only used as a purification process for some bulk product - the impurities to be adsorbed will generally be in the low percentage range. In this application, however, the adsorbing specie will be in the high percentage range. This means that bulk saturation of the adsorbent will occur, with the saturation edge traveling massively downstream in the bed.

Considering the four components of the exhaust gas leads to the conclusion that H_2O should be removed first, then CO_2. Let us look at the H_2O problem first. There are many substances that can adsorb (or absorb) water. A partial list includes silicia gel, activated aluminia, molecular sieves, activated bauxite, calcium chloride, phosphorous pentoxide, and sulfuric acid. The last

three do not deserve consideration here, since they involve chemical processes with considerable disadvantages.

Each of the absorbents listed has advantages, but the best of the lot is the "molecule sieve". This material - a synthetic zeolite - has higher adsorptive compacities than the others, and it can produce lower dew-points at higher temperatures than the others. In addition it has another advantage considered later. To obtain a complete picture of the capabilities of molecular sieves, an engineering analysis would have to be made (References 1, 2 and 3). Adsorption data for these substances are available (Reference 4).

The actual capacity for H_2O depends upon several variables. For the purposes of this discussion it was assumed that sieve No. 4A has a capacity for water of 15 percent of the sieve weight at 25^oC. At 100^oC this would drop to about 10 percent. Using the first figure and the data in column 6 of Table 4-3, we obtain weights of molecular sieve required to take up the gaseous water in the engine exhaust gas.

TABLE 4-5

WEIGHT OF MOLECULAR SIEVE NO. YA

ASSUMING 15% ADSORPTIVE CAPACITY

Reaction	Weight H_2O (grams)	Weight of Sieve (grams)	Volume of Sieve (C.C.)	Heat of Adsorption (K cal.)
1	326	2170	3020	217
2	212	1410	1960	141
3	306	2040	2840	204
4	408	2720	3780	272
5	70	470	655	46
6	254	1700	2360	169
7	164	1100	1530	109

Assuming a density for the sieve of 0.72 g/cc, the volume (column 4, Table 4-5) of sieve required under the assumed conditions is obtained. Since the adsorption process involves the liberation of heat, thought must be given to the effect of this heat and to its removal. For water the heat of adsorption is $\Delta H_{ad} = -0.665$ kcal/gm. Using this value and the data in column 2, the data in column 6, Table 4-5 is obtained.

The picture begins to emerge. To expend 1 kwhr of energy at an engine efficiency of 100 percent would require, for water adsorption, upwards of 3 kg of sieve, 4 liters of sieve volume, and require the removal of 300 kcal of heat. These are not small numbers.

Let us consider the adsorption of CO_2 next. For this purpose another advantage is obtained by the use of molecular sieves. These materials are strong adsorbers of polar molecules. Moreover, they are selective. Sieve No. 4, for example, will adsorb the smaller and more polar H_2O molecule more readily than others. In fact, H_2O could selectively displace other molecules that might be in an adsorbing site. However, since CO_2 is also polar, it will be adsorbed in the absence of water, particularly if a sieve of larger pore size is used. Thus, in a molecular sieve bed the H_2O will be taken up first and then the CO_2.

Data for the capacity of a sieve for CO_2 is somewhat more uncertain than for water. Typical values seem to be about 10-15 percent at 25°C for No. 5A. No value for the heat of adsorption is at hand. Using the lower limit above, data for the weight and volume of absorbent are shown in Table 4-6 for the conditions assumed in Tables 4-3 and 4-4. Density is assumed to be the same as before. It is seen from these figures that an additional 1-4 liters of sieve would be needed to remove the CO_2 per kwh at 100 percent engine efficiency.

TABLE 4-6
WEIGHT OF MOLECULAR SIEVE NO. 5A
ASSUMING 10% ADSORPTIVE CAPACITY

Reaction	Weight of CO_2 (grams)		Weight of Sieve (grams)		Volume of Sieve (c.c.)	
	From Table 4-3	From Table 4-4	On the Basis of Table 4-3 (a)	On the Basis of Table 4-4 (b)	On the Basis of Table 4-3 (a)	On the Basis of Table 4-4 (b)
5	288	160	2880	1600	3860	2220
6	98	78	980	780	1360	1010
7	100	70	1000	700	1390	970

Note:

(a) In Table 4-3 it was assumed that only CO_2 was produced.

(b) In Table 4-4 it was assumed that CO_2 and CO were both produced (see text).

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Chemical Processes

There are many processes for absorbing H_2O that involve chemical or quasi-chemical action. An example would be P_2O_5. However, their application here would have certain disadvantages and probably few advantages. Their use will not be considered.

The usual laboratory method for the removal of CO_2 as an impurity in a bulk gas is to pass the gas through a column containing soda-lime in some form. Usually the active ingredients are supported, in order to increase the active surface. Since carbonates are formed in this process, the recovery of CO_2 from them will probably be done by the application of heat. This heating will, without doubt, destroy the value of the material for reuse.

Unsupported granular lime or sodium hydroxide would be efficient absorbers of CO_2, especially the latter. However, a small amount of water vapor is probably necessary for proper chemical action. With use, these materials would tend to loose their granular form, giving rise to a time-increasing pressure drop across the bed. Furthermore, the use of NaOH would produce water that would cause considerable trouble in the system. CO_2 recovery by subsequent heating would completely destroy any granular properties remaining. The reformed oxides would then have to be reformed by sintering, a procedure which does not appear attractive.

Making use of the weights of CO_2 listed in Table 4-6, it is found that weights of CaO ranging from 90 to 327 grams would be required to take up the CO_2, assuming 100 percent utilization of the CaO. Similarly, 128 to 532 grams of NaOH would be required. To be practical, a use factor of perhaps 4-5 should be assumed, in which case the amount of CaO would be between 450 and 1600 grams per kwhr of energy. This is not significantly better than the molecular sieve and, considering the base process required for recovery, probably considerably worse.

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The use of a chemical process for the task under discussion seems more reasonable and attractive when applied to the removal of nitrogen and carbon monoxide. The gas purification process wherein N_2 is an impurity has long made use of several metals. In principle, any metal that forms a stable nitride can be used to remove N_2. At present these include Ca, Li, Ti, Zr, and U. On a commercial basis Ca has proven expecially attractive. Except for the price, Li would be even more attractive. The reactions are:
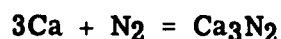
$$3Ca + N_2 = Ca_3N_2$$
$$\Delta H_{298} = -103 \text{ kcal}$$
$$\Delta F_{298} = -88 \text{ kcal}$$
$$3Li + 1/2 \; N_2 = Li_3N$$
$$\Delta H_{298} = -47 \text{ kcal}$$
$$\Delta F_{298} = -50 \text{ kcal}$$
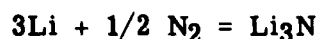
To obtain good kinetics, these reactions are run at high temperatures. In the Ca case a compromise temperature is about 680°C. The use of Li has a further advantage. It may be operated at about 400°C at which point liquid metal is involved. The equipment may be arranged so that fresh liquid surface is continuously being brought into contact with the gas phase.

To obtain a feel for the magnitudes involved, consider the quantities of N_2 produced by reactions 1 to 4 as shown in column 5, Table 4-3. From these values the necessary amounts of Li can be obtained. These data are given in Table 4-7.

TABLE 4-7

WEIGHTS OF Li REQUIRED FOR N_2 TAKEUP

| Reaction	Weight of N_2 (grams)	Weight of Li (grams)
1	126	189
2	124	261
3	158	237
4	106	159

The data in column 3 are again based on one kwh, 100 percent engine efficiency, and 100 percent utilization of the Li. Again assuming a use factor of 4-5, it is seen that about a kilogram of Li metal would be needed.

These metal nitrides cannot reasonably be decomposed by heat. However, base reclamation can be done by the reaction

$$Li_3N + H_2O = NH_3 + 3\ LiOH.$$
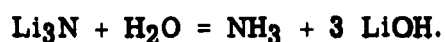
At room temperature this is a highly spontaneous reaction (ΔF_{298}=-191 kcal). The disadvantage is the formation of LiOH, although it can be decomposed by molten electrolysis.

There is a chemical process available for the fixing of the CO. It is not certain that this is commercial, but certainly the reactions can occur. The reactions are
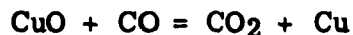
$$CuO + CO = CO_2 + Cu$$
$$H_{298} = -28.0\ kcal$$
$$F_{298} = -30.7\ kcal$$
$$F_{700} = -30.7\ kcal$$
$$2CuO + CO = CO_2\ X\ Cu_2O$$
$$H_{298} = -25.0\ kcal$$
$$F_{298} = -34.5\ kcal$$
$$F_{700} = -36.9\ kcal$$
$$M_{n2}O_3 + CO = CO_2 + 2\ M_nO$$
$$H_{298} = -24.0\ kcal$$
$$F_{248} = -24.5\ kcal$$
$$F_{700} = -27.0\ kcal$$

The first two reactions can go together, since it would be difficult to say whether Cu of Cu_2O alone would be formed. Both would be present. The process, then, would start with supported CuO or $M_{n2}O_3$ (or perhaps M_nO_2). Operation would be in such a manner as not to destroy the physical form of the support. Regeneration of the original reactant would be by reaction with oxygen at the base.

These reactions would require the removal of the CO_2 formed. Table 4-6 shows that this has been taken care of, since the molecular sieve weights and volumes in columns 4 and 6 include these CO_2 values. In this case, then, columns 5 and 7 would be ignored.

Physical Processes

In general, by physical processes we mean those listed below.

1. Absorption of H_2O by solutions or to form solutions. These include H_2SO_4, $CaCl_2$, LiCl, NaOH, glycerin, triethylene glycol, alkali carbonates, and ethanolamines.

2. Condensation or liquifaction.

3. Volume collection.

The substances in the first category are efficient gas purifiers for low percentages of H_2O and CO_2. Their behavior with bulk amounts of gas would need investigation. Probably they could be made to work with suitable arrangement of equipment. The first six are mainly for water although some (NaOH for example) would take out CO_2. This simultaneous removal could cause problems. In general, they are all regenerated by the application of heat.

The ethanolamines have been the subject of a considerable amount of work (Reference 4) over the past thirty years in various industrial processes such as the purification of natural gas. The presence of the amino group in the molecule enables a definite compound to be formed with acidic gases, such as CO_2 and H_2S. Their application would require a tray or packed column. It is difficult to form an estimate of their usefulness at this time.

If the proper equipment were available on a mobile unit, the higher boiling gases could be condensed. It seems reasonable and likely that the H_2O could be rather easily collected as a liquid. Then the traces caused by its vapor pressure could readily be removed, if desirable, by the molecular sieve. This would account for the larger volume of gases in most cases.

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Of the remaining gases - N_2, CO_2, and CO - the CO_2 would be the greatest in quantity. Again, it might be feasible to freeze out the CO_2. If this were not possible, then adsorption by a molecular sieve would not require too much bulk.

The third category consists of simply collecting the exhaust gases in a suitable container, without any deliberate attempt at any other process. The data shown in Tables 4-3 and 4-4 will give an indication of the storage volume needed. For example, reaction 1 would require 508 liters or about 18 cu ft/kwhr. Reaction 6 would require about 17 cu ft. On the other hand if the H_2O were condensed out, reaction 1 and 6 would need about 3.5 and 5.5 cu ft respectively. Should H_2O and CO_2 both be removed, reaction 6 would require 1.75 cu ft.

Let us assume a mission expending power at a rate of 15 kw for a period of 2 hours. Assume further that two times reaction 1 is needed per kwh, and that H_2O is removed from the exhaust gases. The final exhaust would consist of 6120 liters or 220 cu ft of N_2 at S.T.P. This would be equivalent to a sphere 7.5 ft. in diameter. In addition, 6.5 cu ft of molecular sieve would be required to remove the water. Even assuming an additional factor of two on reaction 1 would not create unreasonable volumes. It appears that the collection of exhaust products is not insurmountable.

REFERENCES

1. Barry, H. M. "Fixed Bed Adsorption." Chem. Eng., 67 105 (1960).
2. Perry, J. H. Chemical Engineers Handbook. McGraw-Hill, 1950, pp. 548-555.
3. Kohl and Riesenfeld. Gas Purification. McGraw-Hill, 1960
4. Data sheets from the Linde Company, Division of Union Carbide Corporation.

REGENERATION OF PROPELLANTS FOR RE-USE

Criteria for Selecting a Recycle System

The following paragraphs describe certain desirable characteristics for a lunar-engine propellant regeneration system. These will become more definitive and quantitative as the program develops.

1. The system has been, or can be, demonstrated to perform successfully on earth. Development work would be limited to adaptation and optimization of process and hardware for lunar environment.

2. The system must be relatively free from loss of propellant due to such events as incomplete chemical reaction, formation of undesired by-products, excessive purge requirements, etc. To some extent this loss of material can be evaluated as a weight penalty. However, in general, the preferred system should be free from any need of substantial re-supply from earth.

3. It should be inherently simple to operate and maintain.

4. It should have minimum weight for lift-off.

5. It should have minimum power for regeneration of propellant.

6. It should be compatible with the propellant system used prior to the regeneration stage of lunar program.

Systems

The criteria given above have been applied to regeneration systems for those propellants or fuel-oxidant combinations normally used in industrial, automotive, marine, aircraft and rocket applications which appear to have desirable properties for the lunar engine-fuel system (see Table 4-3). Most of these have been recognized as being unsuited. However, six of the fuel oxidant combinations listed in Table 4-3 have been selected for more detailed investigation, since they definitely meet certain of the criteria.

Preferred Systems

Of the fuel-oxidant combinations listed in Table 4-3, the following are considered promising enough to warrant preliminary design and calculation of power and weight requirements:

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Hydrogen - Oxygen

Ammonia - Oxygen

Methane - Oxygen

Lithium - Hydrogen

Hydrogen - Hydrogen Peroxide

Ammonia - Hydrogen Peroxide

Preliminary data on the first two combinations are included in this report.

The Hydrogen-Oxygen System

This system of regeneration can be considered to start with condensed water in the engine discharge. The water is fed to an electrolysis plant (Figure 4-9) where gaseous hydrogen and oxygen are generated. These gases then are purified and liquefied in a hydrogen liquefier (Figure 4-10) and an oxygen liquefier (Figure 4-11). The liquefied gases are returned to the engine as fuel and oxidant.

The variation of weight with capacity of the electrolysis plant is shown in Figure 4-12. Power consumption (3 kwh/lb water) and heat rejection (1.0 kwh/lb water) are practically constant regardless of plant capacity.

The approximate weight and power curves for the hydrogen liquefier are shown in Figures 4-13, and 4-14, and the corresponding information for the oxygen liquefier is shown in Figures 4-15 and 4-16.

The Ammonia-Oxygen System

This system can be considered as a variation of the hydrogen-oxygen system. The engine exhaust products are water and nitrogen. After electrolysis of the water, hydrogen is stored by fixing it with nitrogen in the ammonia molecule rather than by liquefaction. The ammonia synthesis process is shown on Figure 4-17. Nitrogen and hydrogen gas are passed over a catalyst at high pressure and temperature. The ammonia formed is condensed and removed as product, while the unreacted nitrogen and hydrogen are recycled. The yield is practically 100 percent. The approximate weight and power curves are shown in Figures 4-18 and 4-19.
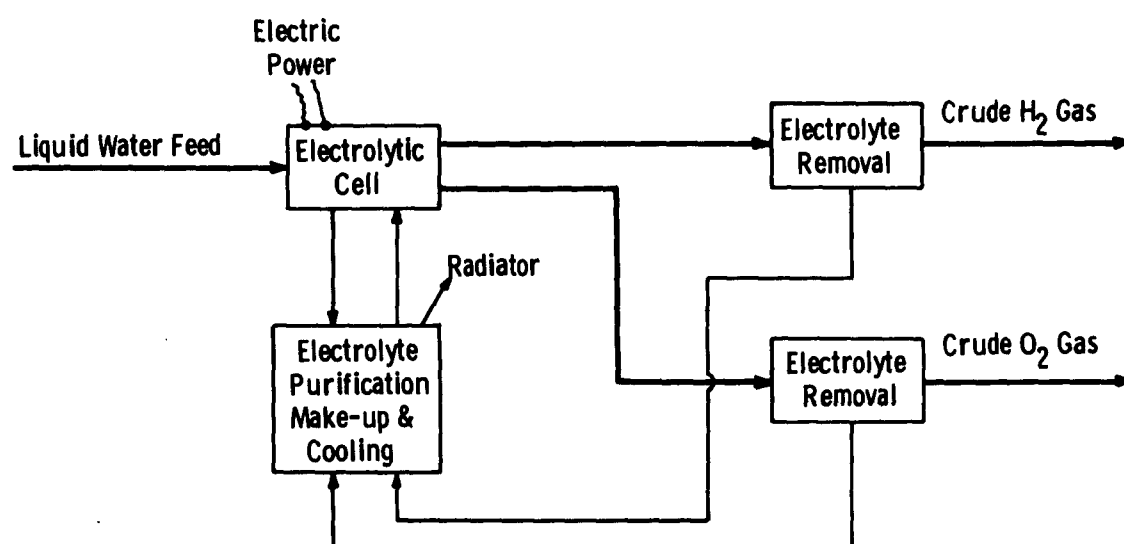
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Figure 4-9. Schematic Process Flow Sheet for Generation of Hydrogen and Oxygen Gases by Electrolysis of Water
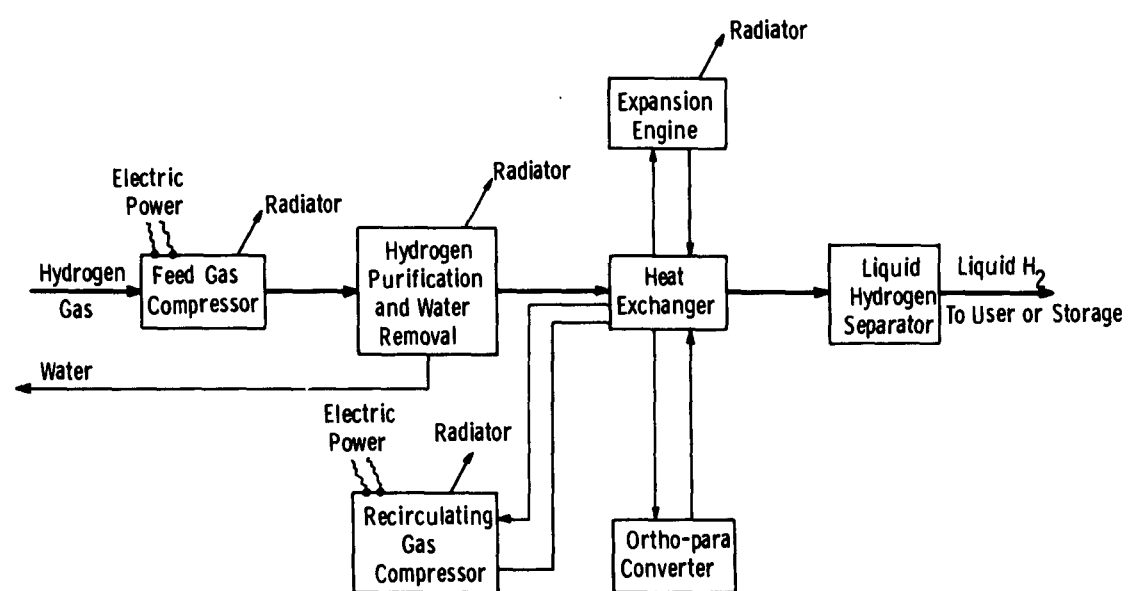
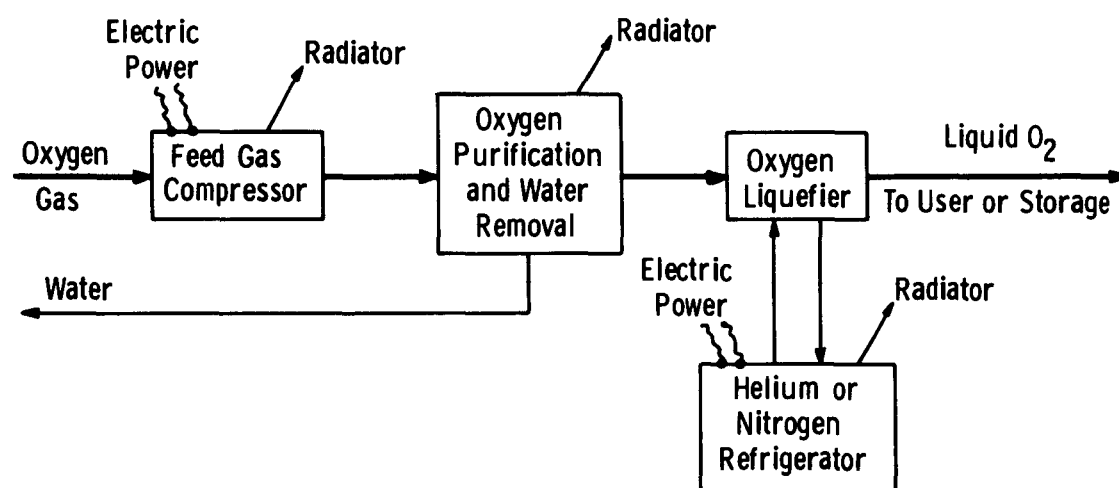Figure 4-10. Schematic Process Flow Sheet for Liquefaction of Hydrogen

Astronuclear

Figure 4-11. Schematic Process Flow Sheet for Liquefaction of Oxygen
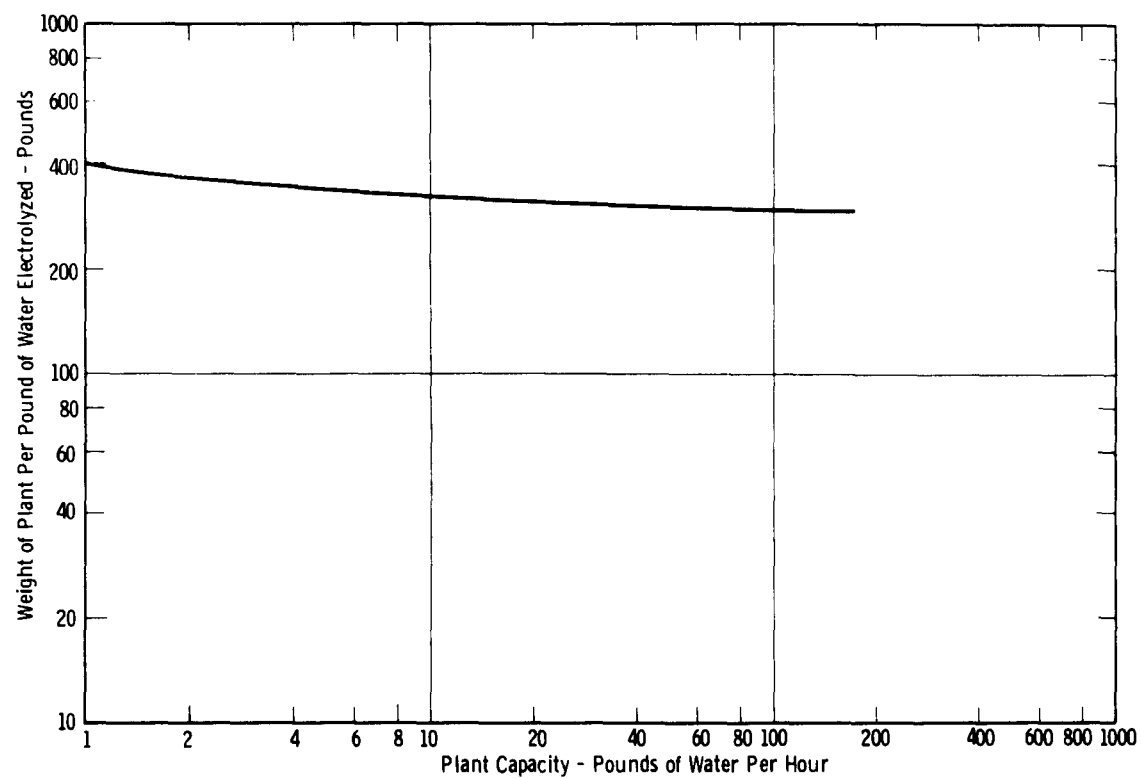
(W) Astronuclear

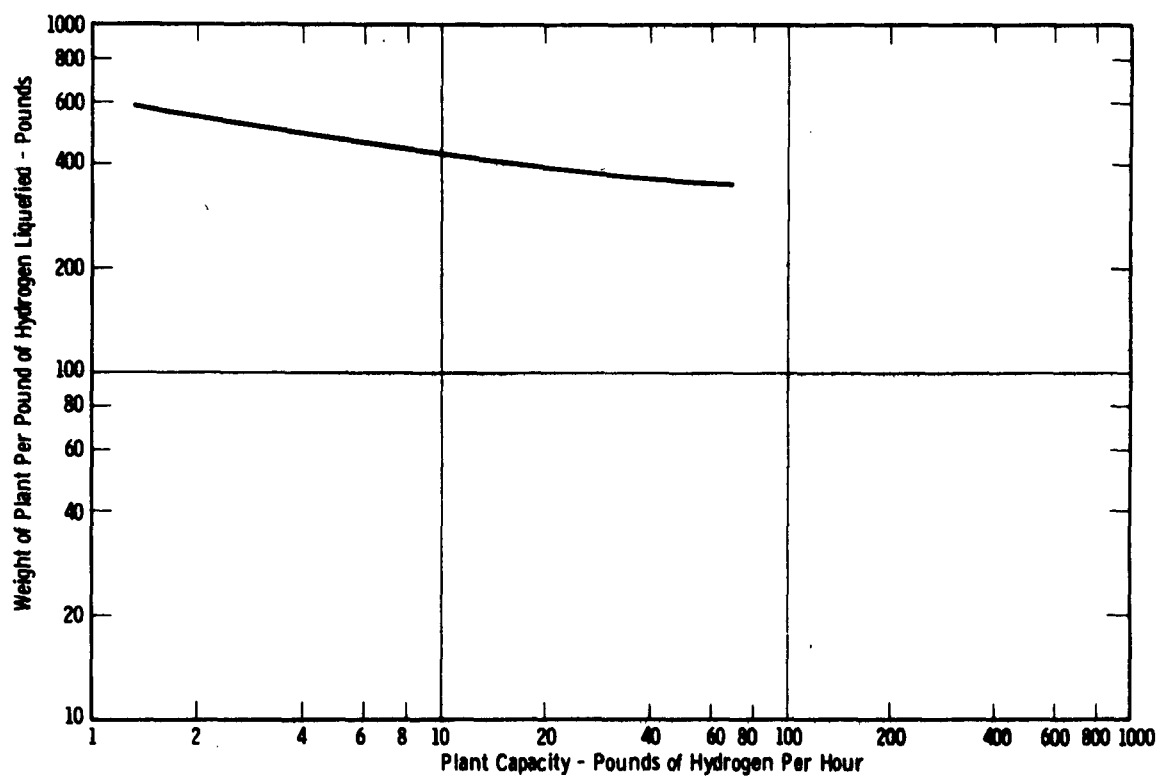Figure 4-12.   Approximate Weight of Water Electrolysis Plant Vs. Plant Capacity

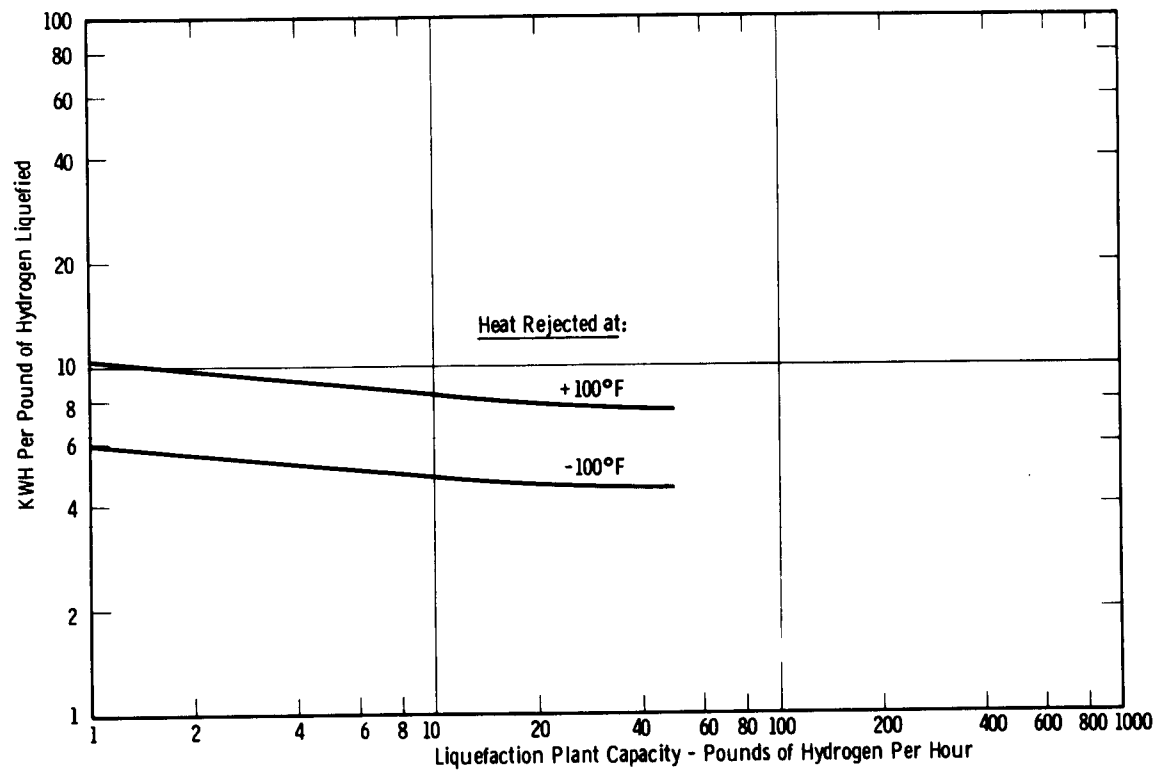Figure 4-13.   Approximate Weight of Hydrogen Liquefaction Plant Vs.
Plant Capacity

Figure 4-14.   Electric Power Requirements for Liquefaction of Hydrogen
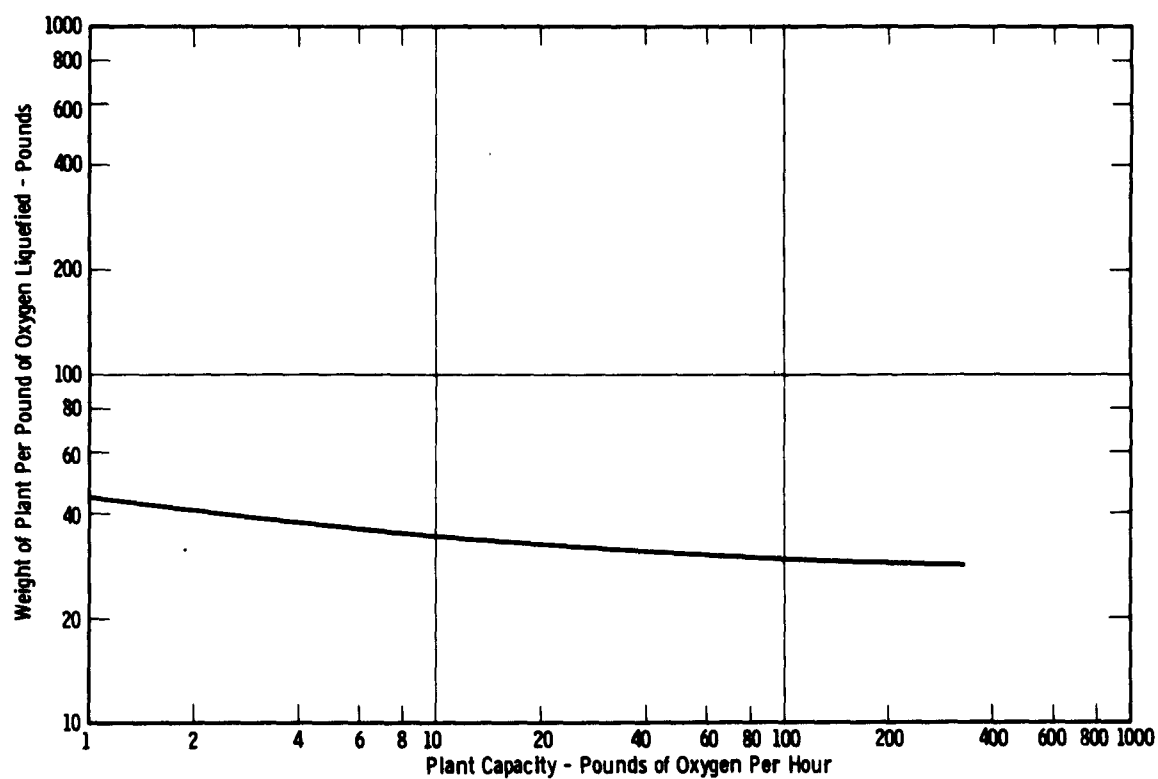
Figure 4-15. Approximate Weight of Oxygen Liquefaction Plant Vs.
Plant Capacity

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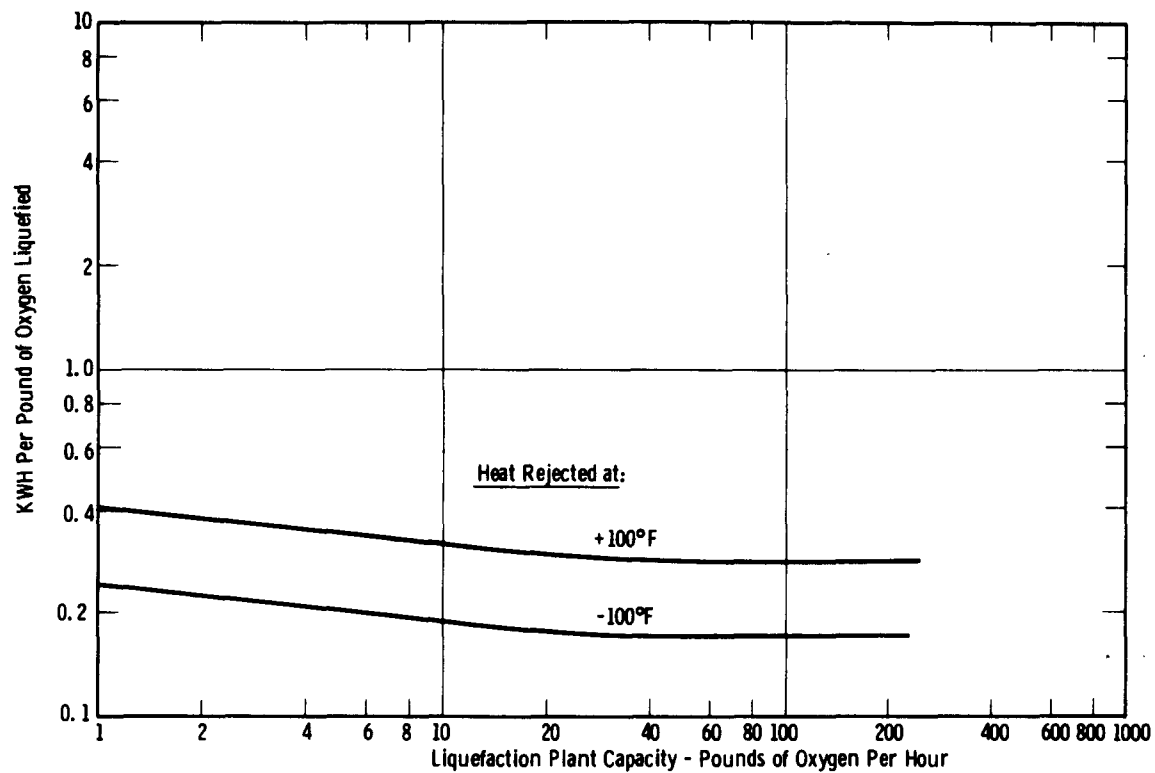
Figure 4-16.   Electric Power Requirements for Liquefaction of Oxygen
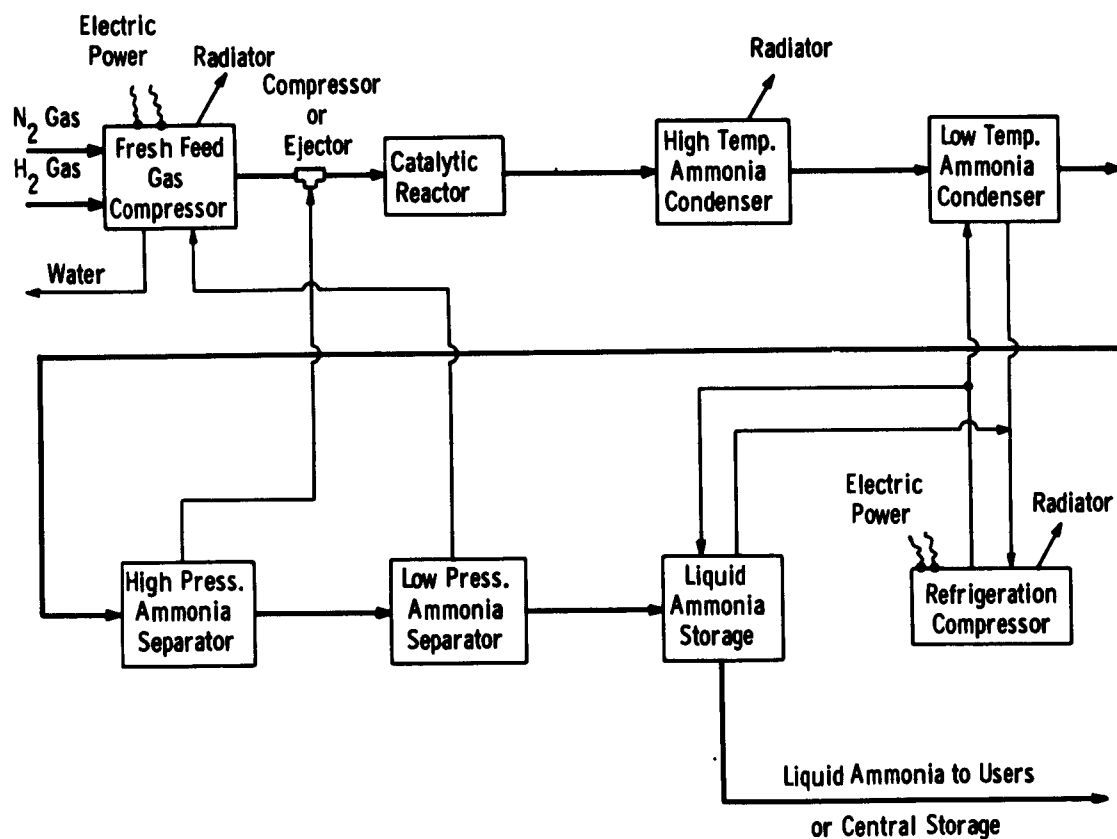
Figure 4-17. Schematic Process Flowsheet for Synthesis of Anhydrous Liquid Ammonia from Nitrogen and Hydrogen Gas
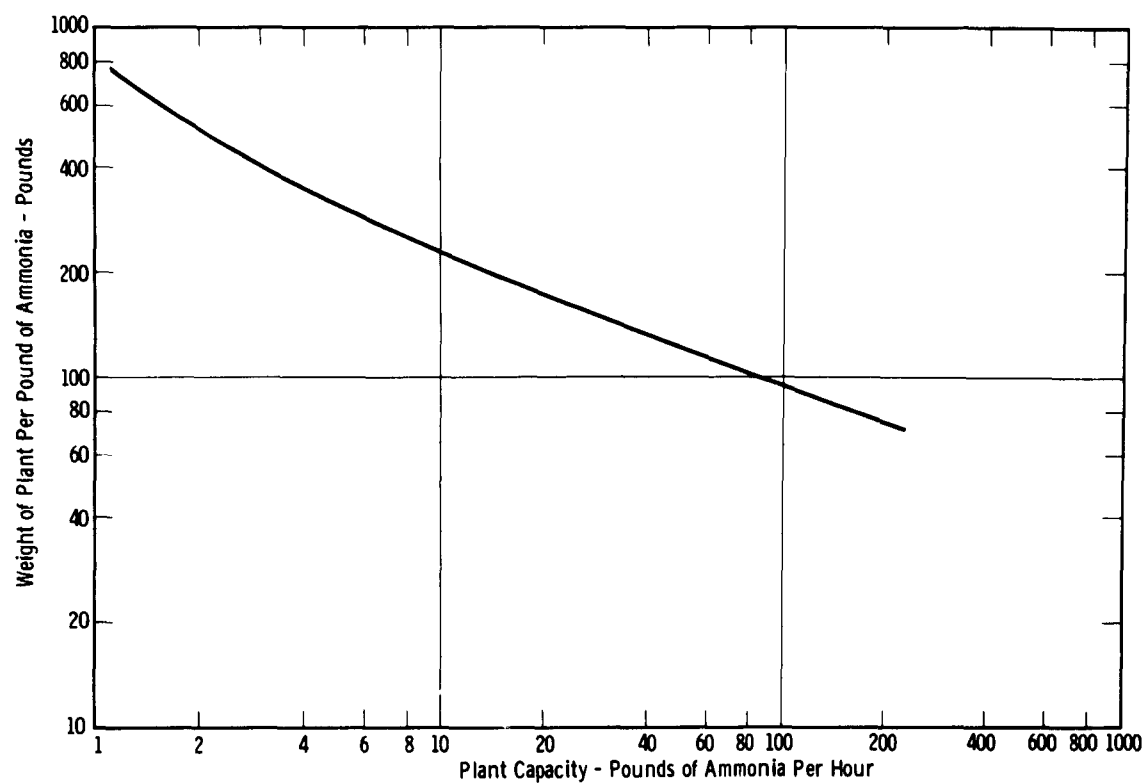
Figure 4-18. Approximate Weight of Ammonia Synthesis Vs. Plant Capacity
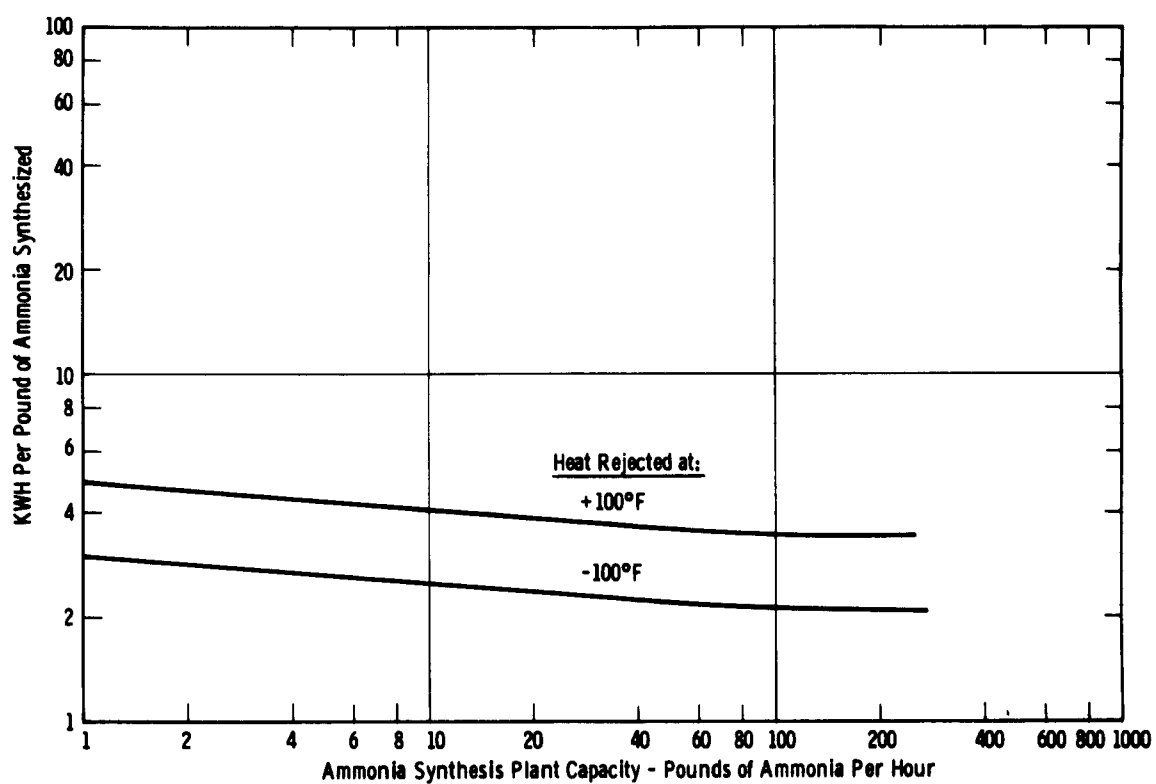
Figure 4-19. Electric Power Requirements for Ammonia Synthesis

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DISCUSSION

Hydrazine and nitrogen tetroxide have a unique position in the propellant
picture, because it is anticipated that there will be substantial quantities of
these materials available on the lunar surface as outage in the LEM vehicle.

This propellant combination does not look attractive for regeneration. The
isentropic SPC of N_2H_4 and N_2O_4 with N_2H_4 as a diluent is approximately
2 times that for H_2 and O_2. Therefore, this combination of fuel and oxidizer
s. ould be considered for use on a non-regenerative basis only if there is a
dependable supply of N_2H_4 and N_2O_4 available.

B. HYDROGEN AS A NON-REACTIVE PROPELLANT

In the Sundstrand cryhocycle engine, which is being considered for the lunar application, the fluid dynamic working fluid, hydrogen, takes the form of a non-reactive propellant. The isentropic SPC for hydrogen in such an engine is shown plotted in Figure 4-20 for the applicable temperature range.

C. SOLAR ENERGY

The value of solar constant for the lunar surface being used in this study is 1.39×10^6 ergs/cm^2/sec (129.3 watts/ft^2)[1].

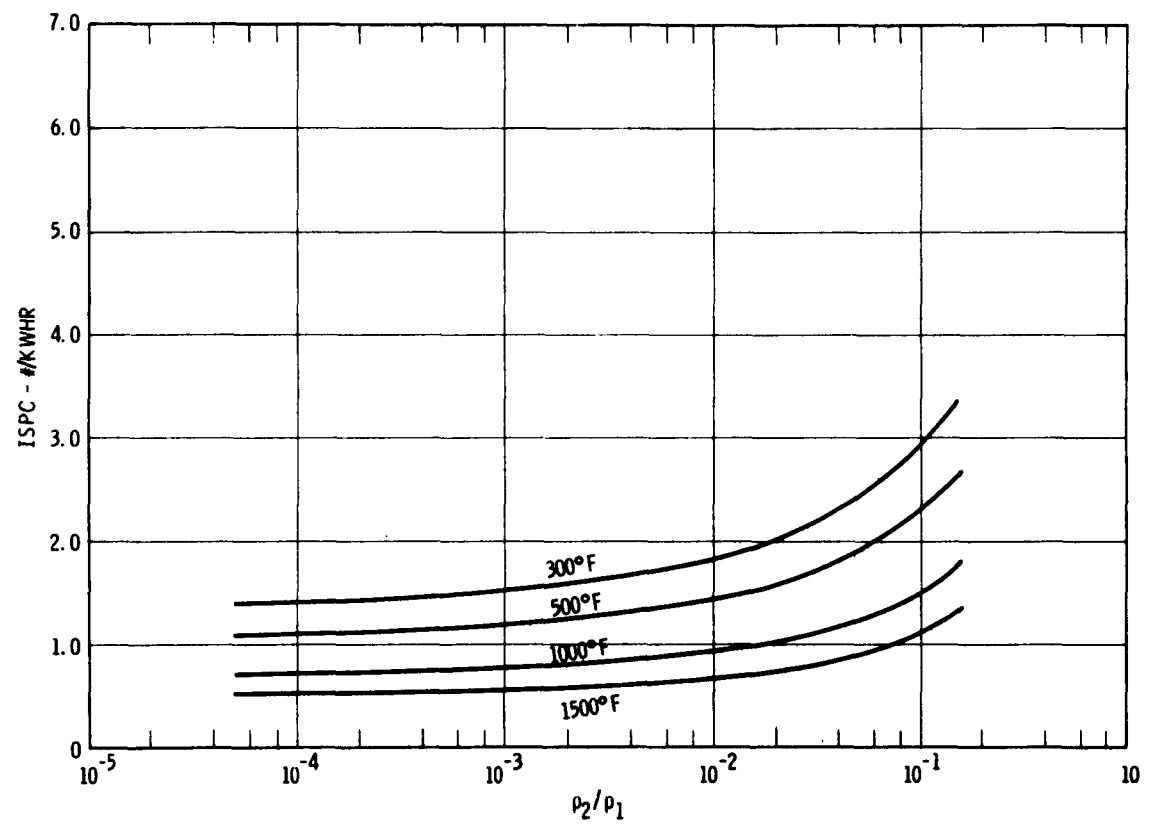1. WANL-TNR-105, Astronuclear Laboratory Space Manual, Part I, The Moon. Westinghouse Electric Corporation

Figure 4-20. Isentropic SPC for Hydrogen

D. RADIOISOTOPE HEAT SOURCES

During the past few years considerable experience has been obtained in the design, development and use of radioisotope powered generators (Reference 1). Recently, the first satellite wholly powered by nuclear energy was put in orbit (Reference 2). This device, known as SNAP-9A, is a plutonium-238 powered thermoelectric generator with an output of 25 watts, weighing 27 pounds and with a design lifetime of 5 years. As with SNAP-9A, essentially all of the experience with radioisotope power sources is in conjunction with thermoelectric generators with relatively low power outputs - a few watts to one kilowatt. This combination appears to offer one of the most reliable and rugged combinations available today. No major difficulty is anticipated in scaling up the design of radioisotope thermoelectric generators to outputs in the range of 1-20 kilowatts. However, the availability of the necessary amounts of suitable isotopes within the next 2-3 years is doubtful.

In Table 4-8 is collected some of the pertinent information on the eight isotopes which appear feasible as power sources now or in the immediate future. Besides availability, other special considerations for this type of source are the shielding requirements and the contamination precautions. Biological shielding, and to a lesser extent, electronic component shielding requirements are rather severe for the beta and beta-gamma emitting isotopes and possibly also for the neutron emitters. On the other hand, the pure alpha emitters require very little shielding. Actual shielding estimates are necessarily dependent on the specific application, taking into consideration not only the particular isotope involved but also the geometry, configuration and distance of the source with respect to the objects to be shielded and the maximum acceptable dosage. It has been estimated (Reference 3) that a strontium-90 powered 4 kw(e) source at a distance of 10 feet from an inhabited vehicle would require about 6000 pounds of shielding to keep the dosage to a limit of 30 rem per year. A similar source powered by polonium-210 would require only about 300 pounds.

TABLE 4-8

RADIOISOTOPE HEAT SOURCES

Isotope	Decay Product	Half Life (Years)	Thermal Power Watts/Gm.	Watts/cc	Chemical Form	Practical Operating Temperature Upper Limit °C	Useful Mission Life Months	Production Capacity 1963 Kw (Th.)/Yr.	1967 Kw (Th.)/Yr.
Cerium CE-144	Beta Gamma	0.78	2.3	13.8	Oxide	2800	9	25.	788.
Cesium Cs-137	Beta Gamma	27.	0.07	0.22	Glass	~1800	>120	5	46
Curium CM-242	Alpha Neutron	0.44	120.	1170.	Oxide		5		9.5
CM-244	Alpha Neutron	18. 1.4×10^7	2.3	22.4	Oxide		>120		41
Plutonium PU-238	Alpha	90.	0.48	9.3	Metal		>120	1.5	11
Polonium PO-210	Alpha	0.38	140.	1320.	Metal		5	3.	14.
Promethium PM-147	Beta	2.6	0.18	1.0	Oxide		31	.01	11
Strontium SR-90	Beta	28.	0.2	0.7	Titanate	~1800	>120	19.	63.

REFERENCES

1. R. T. Carpenter. Space Isotopic Power Systems. Astronautics and Aerospace Engineering, May 1963, pp. 68-72.

2. Chemical and Engineering News, Oct. 7, 1963, p. 37.

3. WANL-TNR-136.

E. WORK PLANNED FOR MONTH OF NOVEMBER

The work planned for the month of November is as follows:

1. Complete thermodynamic analysis of products of listed propellant combinations.
2. Preliminary process design for regeneration of four additional propellant combinations including lithium-hydrogen.
3. Complete compilation of properties of isotopic fuel.
4. Complete preliminary studies of handling and storage requirements for promising propellant combinations.

F. TRIP REPORTS

Reports on trips related to fuels and oxidants are included in this section. Concerns visited included:

1. Battelle Memorial Institute
2. Atlantic Research
3. Callery Chemical Co.
4. Mine Safety Appliance Research Corp
5. Becco Chemical Div., FMC Corp
6. Reaction Motors Div., Thiokol Chemical Co.
7. Arthur D. Little, Inc.
8. Air Products and Chemical Co.
9. Air Research Manufacturing Co., Los Angeles, Div. of Garrett Corp.

BATTELLE MEMORIAL INSTITUTE

Place: Columbus, Ohio

Date: September 24, 1963

Personnel: B.M.I.: Mr. John Clifford, Mr. James Gates.

Subject: Water Electrolysis Units for use in Lunar Environment.

Mr. Clifford is a chemical engineer and the senior man working on electrolysis. Mr. Gates is in charge of a mechanical engineering group and participates in the electrolysis program primarily as the designer and builder of the experimental equipment.

Discussion centered on two electrolysis units on which BMI has done some work: the standard iron-nickel electrode cell of industry, which they have adapted for zero gravity service, and a palladium-silver cathode which they believe to have much promise for generating pure hydrogen.

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Zero-Gravity Unit

The zero-gravity unit is described in detail in "Research on the Electrolysis of Water Under Weightless Conditions", Technical Documentary Report No. MRL-TDR-62-44, May, 1962, Contract No. AF 33(616)-7351. Capacity of the unit was demonstrated at a rate of 4.5 lb. of water electrolyzed per day. One unit has been made for Wright Field and is presently at that location for evaluation.

The individual cells are shallow, circular pans. These are stacked like pancakes to rotate on a vertical shaft. Potassium hydroxide solution used for the electrolyte is held in the cells by centrifugal force. Water make-up is accomplished by means of a small pump in the base of the unit. BMI wired the cells for parallel operation, but they understand that Wright Field is rewiring for series operation.

The weight of the BMI unit is 284 lb. This could be reduced to 200 lb. for any additional units made, since performance has exceeded design specifications. Also, if the rotating feature were eliminated the weight would probably be decreased another 50 lb., so that at the present state of the art a unit modified for lunar service would weigh approximately 150 lb. However, the effect of reduced gravity on cell operation is an unknown factor. BMI feels that the excellent performance they obtained is due, in part, to the high G (over 1.0) caused by rotation. The effect of reduced gravity cannot be predicted and BMI recommends tests in an orbiting satellite.

The unit as built must have the electrolyte removed from the cell before rotation is stopped. This requirement can be eliminated by a minor modification in design.

Heat rejection was not a problem in the unit made by BMI. If it is necessary to cool the cell, as for lunar operation, the electrolytic solution can be circulated through a cooling system.

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Satisfactory operation can be expected at a current density of 100 amperes per sq ft. Approximately 1400 watts are required to electrolyze 0.5 lb. of H_2O per hour.

Water purity has not been a problem in test operation of the BMI unit. In the proposed recycle operation, purification of water feed might be required.

Palladium-Silver Alloy Cathode

Some beaker-scale work has been done with a palladium-silver alloy cathode. Considerable further work would be necessary before any worthwhile design data could be obtained. However, the use of this electrode offers very interesting possibilities for the design of a more efficient and more compact, lighter cell. Significant features of a cell utilizing this electrode material are:

1. An appreciably lower cell voltage is possible--a range of 1.4 - 1.7 appears quite feasible.
2. Current densities of the order of 900 amp/sq ft have been used, and still higher densities are expected to be possible.
3. Hydrogen does not bubble off the electrode but diffuses through it and is collected as high purity, dry hydrogen. One of the principle difficulties of this cell is to maintain the electrode poison-free and the hydrogen transmission at essentially 100 percent. To date, electrodes have been operated continuously for more than 70 hours in a satisfactory condition.

The present tentative estimate is that a unit with a capacity of 6.7 lb. of water/day (three-man unit) could be built which would weigh about 25 lb. and occupy about 0.25 cu ft.

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ATLANTIC RESEARCH CORPORATION
Alexandria, Virginia

September 25, 1963

Those present: Atlantic Research: Tom Perrelli, Preliminary Design and Proposals; Dr. Jack Godfrey, Propellant Research Dept.; Bernie Silver, Project Engineer on Gas Producers; Bill Sargent, Propellant Research Dept.

Atlantic Research was founded in 1949 and has grown into a $35,000,000 corporation. It originally did contract research work, but has expanded into development and production. It is primarily a solid propellant manufacturer and has pioneered in the use of aluminum and beryllium in solid propellants.

Solid propellant rockets manufactured by Atlantic Research have been used for small missiles, for vector control of large missiles and spacecraft, and for sounding rockets. Modified solid propellants have been used for gas generators.

Several years ago ARC initiated gelled propellants as an expedient for testing experimental solid propellants. These gelled propellants are the same formulations as the solid propellants in a thixotropic form and have handling properties which approach those of liquids. ARC has worked out a large rocket configuration using the gelled propellant which has some important advantages over standard solid propellant rockets.

The main objective in visiting ARC was to obtain information on their propellant materials which have been used as gas generators. Up to this time they have used solid propellants in the gas producing devices they have made.

There are two categories of gas producer: clean and dirty. The specifications for a clean gas generator are that the solids be less than 0.5 percent and the particle size be less than 50 microns. The solid gas producer materials made by ARC are of the clean type.

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There are two serious disadvantages inherent in the use of solid propellant gas producers in the lunar application. First, the reaction cannot be modulated and if the load varies the only means of control is dumping. This, of course, would give a high propellant consumption. Second, it is difficult to get solid propellant grains which will burn for long periods of time. The burning rate normal to the reaction interface is about 0.04 in/sec., which means that an end-burning grain to last 30 minutes would have to be 72 in. long. In order to get continuous operation with solid propellants, it would be necessary to develop a mechanical method of sequencing charges.

The longest burning time that any of their gas producers has had is five minutes. It was found that the rocket casing temperature stabilized after 70 seconds.

The solid propellant that they are using for gas producing contains ammonium perchlorate as an oxidizer and plasticized PVC as a fuel and binder along with numerous other materials in small quantities. The flame temperature of this material is between 1900 and 2000°F.

ARC thinks that it is feasible to make this same formulation in a gel or thixotropic form and also to modify the formulation to obtain a gas temperature in the order of 1500°F. HC1 could be eliminated from the products of combustion by replacing the perchlorate with nitrate.

ARC has done considerable work in atomizing and injection of the gelled material and has a reactor designed for it which has a high turn down ratio and a clean cut-off. The gel can be ignited hypergolically.

The gelled propellants have been stored for up to four years in polyethylene bags without any decomposition or separation. Temperatures were cycled over a few hundred degrees range during this period. ARC has developed methods of hermetically storing their propellants.

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ARC is going to send us complete information on their present solid gas generator propellants and their best opinions on what can be done in developing a gelled gas generator propellant for the lunar application. ARC suggested that we look at solid O_2 generators as a intermediate O_2 storage medium. We were told that Olin Mathieson and Standard Oil of Indiana are their major competitors in solid gas generator propellants. Bell Aerosystems was suggested as a good source of information on storable liquid propellants.

CALLERY CHEMICAL COMPANY

Place: Callery, Pa.

Date: Sept. 26, 1963

Personnel: L. J. Edwards, H. S. Uchida

Subject: Special Fuels and Oxidants

Oxidants

Fluorine and various fluorine bearing compounds were discussed briefly. All of these are difficult to store and handle, are health hazards, and give reaction products which are quite corrosive. Oxygen difluoride has a small energy advantage (maximum specific impulse) with most fuels and also a weight (density) advantage over other fluorine compounds and fluorine.

Only one oxygen bearing compound, nitronium perchlorate, NO_2ClO_4, was discussed in detail. This is said to be one of the most practical oxidants presently available. It is a solid, quite stable below about 90°C when not in contact with moisture or certain organic materials. More oxygen is stored per unit volume in nitronium perchlorate than in liquid oxygen. Nitronium perchlorate can be decomposed without burning or explosion by heating to about 125°C.

Fuels

The properties of various boron compounds were discussed. Of particular interest are the systems composed of the various boranes and hydrazine. These systems require no other oxidant; the products of combustion are boron nitride and hydrogen. They have a much lower flame temperature (about $2500^{\circ}K$) than other systems of comparable specific impulse. The system pentaborane-hydrazine (B_5H_9 - N_2H_2) is said to be the best storable system known (the boiling points of the two components are $60^{\circ}C$ and $113^{\circ}C$ respectively).

The most recently developed boron fuels are QMB-3, (tetramethyl ammonium triboro hydride, $(CH_3)_4NB_3H_8$) and ammonium triboro hydride, $NH_4B_3H_8$. Both of these are relatively stable solids and provide better storage of hydrogen per unit volume than either liquid or solid hydrogen. More information on QMB-3 can be obtained from Mr. J. Ardinger, Bureau of Weapons, RMMP-2.

MSA RESEARCH CORPORATION

Place: Callery, Pa.

Personnel: Robert C. Werner, Thomas A. Ciarlariello

Subject: Lithium Hydride

The properties of lithium hydride, the use of lithium hydride as an energy storage medium, and the lithium-hydrogen fuel cell were discussed. A compilation of the physical properties of lithium hydride was obtained in the report, MSAR 62-59, Physical Properties of Lithium Hydride by T. A. Ciarlariello.

A proposed energy storage-energy conversion device based on lithium hydride and the lithium-hydrogen fuel cell was described. This is a heat to power conversion cycle and the overall thermodynamic efficiency is limited by the

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Carnot cycle efficiency. Based on a fuel cell temperature of 350°C and re-generator temperatures of 900°C for regenerating and 250°C for storage, an efficiency of about 13.8 percent and a 48 hour storage capacity is claimed.

The system described by Zachmann ("Integrated Environmental Control, Power Supply and Propulsion Equipment for a Manned Lunar Surface Vehicle" by H. Zachmann, The Martin Co., Report No. 2726-62) consisting of a hydrogen expansion engine, silver-cadmium battery, lithium hydride fuel cell, and nu-clear reactor was recommended for further study.

The MSA representatives implied that they had information on somewhat more sophisticated lithium hydride systems including data on efficiencies, weight, etc. which they considered proprietary at the present time.

INORGANIC CHEMICAL DIVISION OF FMC CORPORATION
Buffalo, New York

October 3, 1963

Those present: FMC Corporation: James C. McCormick, Technical Sales; George Swinski, Process Engineer.

The Inorganic Chemical Division of FMC Corporation was formerly the Buffalo Electro-Chemical Co. (BECCO). The Buffalo plant is now solely a production plant and produces mainly hydrogen peroxide. They make hydrogen peroxide in concentrations varying from 27-1/2 to virtually 100 percent.

FMC Corporation has a research laboratory at Princeton, New Jersey. The executive headquarters of the corporation are in San Jose, California, and those of the chemical operation are in New York.

Mr. McCormick is on the New York technical staff, but is in residence at the Buffalo plant. He has wide experience in the application of hydrogen peroxide in the military and space fields and has served as a consultant to NASA and several space firms in the application of hydrogen peroxide.

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The following uses of hydrogen peroxide in the space field have been made or proposed:

1. Source of O_2 and H_2O for life support.
2. Oxidant in liquid or hybrid bi-propellant propulsion systems.
3. Monopropellant for driving liquid reactant pump turbines in propulsion systems.
4. Monopropellant for driving APU turbines.
5. Monopropellant for small vector control rockets.
6. Radiation shield for personnel.
7. Personnel heating.

A list of the material given to the writers by Mr. McCormick on the properties and uses of hydrogen peroxide is attached.

For the purpose of engine-fuel system study, the use of hydrogen peroxide as a monopropellant to produce gas to drive a turbine or reciprocator and/or as an oxidant in a bi-propellant internal or external combustion system is feasible. Reference 8 states that hydrogen peroxide can be stored at ambient temperatures for periods up to three years in aluminum alloy drums with very little oxygen loss. The loss is increased significantly at temperatures of about 200°F. The higher the H_2O_2 concentration, the greater the stability. Reference 9 shows that highly concentrated H_2O_2 solutions have good compatibility with many materials of construction.

Reference 7 states that hydrogen peroxide was not decomposed significantly when exposed to radiation in a reactor core. Mr. McCormick said that based on these tests, it might be feasible to regenerate H_2O_2 by using radiation from the nuclear plant reactors. (Dr. Sun will be consulted regarding this.) The decomposition temperature of hydrogen peroxide ranges up to about 1800°F for 100 percent concentration. This is about the same range of temperatures covered by current gas turbine technology. Therefore, Brayton

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cycle turbine design technology could be applied to a turbine application using decomposed H_2O_2 unless there is a substantial effect of the increased O_2 content on the material properties.

References 5, 12, and 13 show that hydrogen peroxide can be used as an oxidizer with any of the fuels used in rocket bi-propellant systems and will give energy releases comparable with those obtained with oxygen and fluorine. Reference 1 shows that hydrogen peroxide can be used satisfactorily with hydrocarbon fuels.

FMC produces hydrogen peroxide by two different processes; i.e., electrolytic and organic. The electrolytic is more costly, but gives a purer product. The electrolytic process is used for producing the higher concentrations which are generally used only in space or military applications. The basic electrolytic process produces a concentration of about 30 percent. Higher concentrations are made by distillation.

Information on the electrolytic process has been published in the technical press (see Chemical Engineering for August 1954, pp. 304-307). Mr. Swinski stated that the information given in this paper is up to date and reasonably complete.

FMC ships and handles large quantities of the concentrated solutions of H_2O_2 very much like one handles water. Once a railroad or truck tank is conditioned for H_2O_2, it can be reused repeatedly without special treatment. The drums used for smaller quantities do have special vents and provisions for preventing spillage. The drums are also cleaned after each use.

Visit to Inorganic Chemical Div.

Properties and Uses of Hydrogen Peroxide (from FMC)

1. H_2O_2-Diesel Cycle.
2. Thermodynamic Properties of 70 thru 100 percent H_2O_2.
3. Article on Hyprox Steam Generator.

4. Bulletin No. 104: Materials of Construction for Equipment in Use with Hydrogen Peroxide.

5. Rocket Propellant Combinations Using Hydrogen Peroxide.

6. Bulletin 67: Hydrogen Peroxide Physical Properties Data Book.

7. Technical Data: Effect of Radiation upon 98 percent H_2O_2 and Water.

8. Technical Data: Excellent Storage Record with Becco 90 percent and 98 percent H_2O_2.

9. Technical Data: The Excellent Compatibility of Materials of Construction with Becco 90 percent and 98 percent H_2O_2.

10. Drawing: 90 percent H_2O_2 Personnel Heating System to Supply 800 Btu/ hour.

11. Bulletin 112: Use of Concentrated Hydrogen Peroxide for Atmosphere Control in Space Flight.

12. Reprint: For All-Around Propellant Performance: 98 percent H_2O_2, James C. McCormick, Space/Aeronautics, March 1963.

13. Bulletin 107: High Strength Hydrogen Peroxide Monopropellant and Bi-propellant Data.

14. Mollier Diagrams for Products of Decomposition of 70 percent, 80 per-cent, and 90 percent Concentration by Weight of Hydrogen Peroxide.

15. Properties of 65 thru 98 percent H_2O_2.

REACTION MOTORS DIVISION OF THIOKOL CHEMICAL CORPORATION
Denville, New Jersey

October 9, 1963

<u>Those present:</u> Reaction Motors: William V. Chambers, Manager, Research Marketing; Dr. Stan Tannebaum, Chemistry Group, Research Dept.; Dr. John Paustin, Chemistry Group, Research Dept.

Reaction Motors has been primarily a liquid propellant manufacturer. Since becoming part of Thiokol, it has gotten into some solid work. Their current

work in liquid propellants mainly involves monomethyl-hydrazine as a fuel and oxygen difluoride as an oxidizer.

Reaction Motors has a line of gelled or thixotropic heterogeneous monopropellants which contain ground carbon particles as the fuel component. These materials do not detonate because the particle size is kept above 20 microns. Red fuming nitric acid and nitrogen tetroxide are used as oxidizers. Boron carbide can be substituted for the carbon. These materials decompose slowly at elevated temperatures and have a flame temperature in the order of 3000°K. The products are quite clean and most of the nitrogen is N_2. These materials are ignited by UDMH.

Work on the gelled heterogeneous propellant is being done under Contract NOw-63-0396C. Information on this material has been published in the Bulletin of the Fourth Joint Army-Navy-Air Force-NASA-ARPA Liquid Propulsion Symposium - 6-8th November 1962, LPS-62-1 (classified).

Two solid monopropellants were discussed. The first is a heterogeneous mixture of hydrazine and diborane and is in the conventional solid grain form. The second is a homogeneous material called hydrazine azide ($N_2H_4 \cdot HN_3$) which is a crystalline solid having a melting point of 84°C. Hydrazine azide decomposes to N_2, H_2, and a small amount NH_3 at a temperature of about 150°C. The isobaric flame temperature is about 1790°K. Hydrazine azide has good stability and is not highly shock sensitive, has a density of 1.39 gm/ml, and has a heat of formation of +55 heat/mol. It is synthesized by neutralization of hydrazoic acid (HN_3) with hydrazine (N_2H_4).

Dr. Tannenbaum has done some work on combinations of borane compounds and hydrazine or ammonia. (Note: Use of such materials had been suggested to Dr. Snyder and Dr. Ashcraft by Callery Chemical Co.) He has found that pentaborane and hydrazine are the best combination. Rocketdyne has a contract to develop a rocket engine to use these materials and has found that

they are unreliably hypergolic and are difficult to get to react completely. Rocketdyne has been able to get combustion efficiencies of 90 percent after considerable development effort.

Dr. Tannebaum stated that he thought that the other boron compounds with hydrazine or ammonia did not have good operating characteristics.

A. D. LITTLE, INC.
Cambridge, Mass.

Date: October 15, 1963

Place: Cambridge, Massachusetts

A. D. Little Personnel: Herbert H. Howell, V. P. Marketing; Dr. Peter Glaser, Mr. Aubrey Tobey, Mr. Arthur Fowle, Mr. Carl Walker, Mr. Charles Schulte, Dr. Ralph Horn, Mr. James George.

A. D. Little maintains a strong interest in the fundamental problems relating to cryogenic processes and hardware. Considerable effort is expended in the development and testing of liquefaction equipment and insulation for cryogenic storage.

Liquefaction Equipment

ADL's capabilities in helium liquefiers are well known, since they have made and sold a number of these units in past years. At present, they are completing the assembly of a unit which will liquefy approximately 100 liters per hour.

Of specific interest to the lunar program is a combination compressor-expander currently being tested and developed. This is a high speed reciprocating machine with opposed pistons, one piston being used for compression, and the other for expansion.

A brief inspection was made of the various liquefaction units under test or in manufacture in the ADL shop.

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Astronuclear

It is the opinion of ADL that within approximately three years, liquefaction apparatus capable of meeting the lunar program criteria could be developed from the present state-of-the-art.

Insulation

An inspection was made of the equipment for measuring the heat transmission of laminar reflective type insulation. There was discussion of both theory and details concerning the design and use of insulations for cryogenic storage.

In ADL's opinion the present state of the art on cryogenic storage vessels would probably limit the size of "no-loss" cryogenic tankage to approximately 10 ft diameter vessels. They feel that a reliquefier may be more reliable than a "no-loss" insulated vessel.

Miscellaneous

An inspection was made of the electrochemical laboratory. There was a general discussion of fuel cells and batteries. The opinion was expressed that fuel cells were approaching 12-18 month reliability.

AIR PRODUCTS & CHEMICALS CORPORATION
Allentown, Pa.

October 10, 1963

Those Present: Air Products & Chemical Corporation: Walter Snow, Advanced Products Dept., Mgr; Richard Clark, R & D Project Mgr, Military Engine Fuel Depot; George Siegrist, Technical Specialist in Advanced Projects Development, Defense & Space Division; Robert Barclay, Project Mgr, R & D; George Schmauch, Section Mgr, R & D; John Carlson, Mgr, Advanced Projects Development, Defense & Space Division; Kenneth Zeitz, Marketing Mgr, Defense & Space Division.

Air Products & Chemical Corporation (formerly Air Products Incorporated) was visited to determine the part they are playing in space application and design of cryogenic storage and liquefaction systems.

In the area of storage tanks, a limited amount of design and construction work has been attempted by AP & C. They have designed and built standard cryogenic storage vessels for earth use. One study for space storage involved a 10 ft diameter x 17 ft long tank for liquid hydrogen in an earth orbit. This was a special design and entirely theoretical, no work having been done on actual fabrication and handling techniques.

Their experience and capability in the liquefier field is fairly extensive. They have designed and tested several miniature refrigeration units for infrared coolers in the 5-20°K range. These units however are generally designed for miniaturization rather than low weight or power. They are testing two small specially designed compressors for space application in the 0.3 to 6 kw power input range. These units are in a fairly well developed stage, and could be incorporated in a liquefaction system without much development work. We observed the testing of a small expansion engine to operate in the low flow range. This engine however, has not demonstrated the reliability necessary for lunar operation, its present life being approximately 200 hours.

Some information was presented that would be directly applicable to the lunar study. This was a review and analysis of a paper by G. Seigrist which contained parametric data for determining the weight and power requirements for a 2 lb/hr hydrogen liquefier. These figures were based on component operational data, extrapolated data, and theoretical considerations.

On a tour of the research laboratories, we were shown the various government and company funded projects now in progress.

These projects consist mainly of determining physical properties of various substances at low temperature. A few of the projects being worked on are: boiling characteristics of liquid neon, thermodynamic properties of gaseous mixtures, and thermal conductivities of various insulation.

Astronuclear

AIR RESEARCH MANUFACTURING CO. A DIVISION OF THE GARRETT CORP.
Los Angeles, California

October 15, 1963

Those Present: For Air Research: R. S. Hunt, Project Administrator, Sales
Dept.; Dr. Holmes, Consultant; George Kunis, Research Dept.; J. J. Kennedy,
Cryogenic Systems, Engineering; K. F. Jackson, Space Environmental Controls, Engineering.

General

Air Research was one of the bidders on the lunar engine-fuel system and is
working with Boeing on their lunar base study in the area of life support.
Air Research's Phoenix Division is prime for the Spur contract.

The Garrett Corporation and its Air Research Divisions are engaged in the
following space related programs:

1. Closed Brayton cycle engines
2. Rankine cycle engines
3. Sterling cycle engines
4. Wheel drive system for earth vehicle
5. Miniaturized cryogenic systems
6. Refrigeration systems for temperature as low as that of liquid helium
7. Cryogenic storage
8. Life support systems
9. Electrolysis of water

Electrolysis of Water (Mr. George Kunis)

Air Research is doing research on a high current density electrolysis system
which they expect to have a current density of about 2000 amperes/ft at 2-2.3
volts. This system is suitable for use in zero gravity. No information was
given as the operating principle of this electrolysis cell.

Life Support

Air Research is not convinced that the life support system should have a diluent gas. Howevever, if a diluent gas is used, it should be nitrogen.

Oxygen consumption in a pressure suit is expected to be about 2 times the normal amount. The ability of a man to do work in a pressure suit is very low. The maximum walking rate in a suit is about 0.4 mph.

High Pressure Storage of Gases

Gas storage is preferred for only very small quantities, e.g. 5 lb of O_2. Air Research is working on filament wound fibreglass high pressure tanks. Some of the tanks look very good, but it is difficult to get repeatability in the physical properties of the material.

The optimum storage pressure for gases is about 2000 psi.

Cryogenic Storage of Gases

Air Research has the contract for cryogenic storage of gases for the Gemini program.

They think that sub-critical vented storage vessels are probably best. The largest tank which is practical is best, and a spherical tank has the lowest weight penalty. Long term cryogenic storage of O_2 is fairly easy, but hydrogen is a problem because the low density gives a high surface to weight ratio. Air Research's present practice is to sub-cool to 2 psia.

Passive reliquefaction during the lunar night is promising for oxygen but not for hydrogen. The oxygen would be in a nitrogen shielded container with passive regeneration of the nitrogen during the lunar night.

Turbo-machining type refrigeration systems are being developed by Air Research for liquefaction of neon, nitrogen, and helium using the Claude by-pass principle. The rotative speeds of these units range from 200,000 to 400,000 rpm. Ways have been found to reduce windage losses to a few percent. Power requirements range from 2-1/2 to 5 watts/100 lbs.

Air Research has found that the thermal conductivity of the super insulating materials attainable in field installation is an order of magnitude greater than that obtained in laboratory tests by the manufacturer. They consider NRC super insulation superior to that made by Linde. There are two problem areas in the application of super insulation:

1. Cutting or folding to fit surface
2. Filling tubes and support pads

The materials used are Inconel 718 for oxygen and titanium for hydrogen. Maraged steels look good for both gases and can be hydroformed and deep drawn. The largest tank which they have made had a 47 in. inside diameter.

Air Research has made slush hydrogen by evaporative cooling. Handling of the slush is a problem, especially the metering. They consider the technology to be at a state where it could be reduced to practice with 6 months of development effort.

Air Research has a computer program for optimizing the design of cryogenic tankage as regards thermal and mechanical characteristics. It was stated that Air Research would probably be willing to provide a design service for use. Mr. Hunt was requested to give us an estimated cost of using the program.